AEFA Project No. 87-25-1

EVALUATION OF THE IMPROVED OV-1D ANTI-ICING SYSTEM PHASE II

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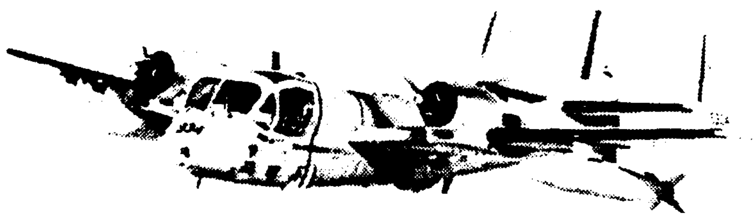

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The U.S. Army Aviation Engineering Flight Activity re-evaluated the OV-1 modified improved anti-icing system from 6 to 23 March 1989, at Duluth, Minnesota. Six flights (6.4 hours of cloud immersion) in natural icing conditions were conducted. Moderate icing conditions with temperatures less than -12 degrees C were not encountered. One enhancing characteristic, seven deficiencies and five shortcomings were noted. The windshield anti-ice system enhances safe mission accomplishment. The seven deficiencies are as follows: (1) the failure of the pneumatic deicing system to remove wing and empennage leading edge ice accumulations in moderate icing colder than -10 degrees C; (2) the failure of the anti-icing system to keep the engine inlet clear at temperatures colder than -10 degrees C; (3) the ice accretion and shedding characteristics of the propeller spinner afterbody which may result in ice foreign object damage (FOD) to the engine; (4) the numerous converter dropouts during normal operation; (5) the ice accretion characteristics of the pitot tube which result in erroneous airspeed indications; (6) intermittent failure of the ice detector system to activate the anti-icing system; and (7) the actions required to re-establish flight-essential and normal inverter loads following inverter dropout. The shortcomings are as follows: (1) the airframe damage caused by the shedding characteristics of the propellers; (2) the propeller damage caused by the ice accretion and shedding characteristics of the windshield wipers; (3) the illumination of the Master Caution Light during anti-ice system activation; (4) the reverse sensing

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and mislabeled engine anti–ice switch; and (5) the occasional dropout of the windshield anti–ice system and the lack of failure warning to the pilot. The OV–1 aircraft with the improved anti–icing system should be cleared into icing conditions up to and including moderate with ambient temperatures no colder than -10 degrees C. Engine acceleration tests were conducted up to 25,000 feet pressure altitude with one engine's AC generator carrying the full anti–ice/deice electrical load. No limits were exceeded, however increased acceleration times were expected and noted.

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INTRODUCTION

BACKGROUND

1. The U.S. Army Aviation Engineering Flight Activity (AEFA) previously identified two deficiencies and seven shortcomings with the operational OV/RV-1D anti-icing system (refs 1 and 2, app A). The deficiencies were as follows: (1) ice accretion characteristics of the engine cowling, propeller, and propeller spinner can result in engine damage and (2) failure of the windshield ice protection system to clear the windshield sufficiently to provide adequate forward visibility after encountering icing conditions. Grumman Aircraft Corporation (GAC) developed an improved anti-icing system for the OV-1 Aircraft. AEFA evaluated the improved system in artificial and natural icing conditions during the 1987-1988 icing season (AEFA Project No. 87-25). One enhancing characteristic, nine deficiencies, and three shortcomings were identified and reported (ref 3). Due to numerous system component failures and dropouts during Project No 87-25, it was not possible to conduct a valid test of the system in icing conditions. The U.S. Army Aviation Systems Command (AVSCOM) contracted with GAC to develop modifications for the improved anti-icing system and tasked AEFA to evaluate the modified system during the 1988-1989 icing season (ref 4). Testing was conducted in accordance with the approved test plan (ref 5). Flight restrictions and limitations provided by the modified operator's manual (ref 6) and the airworthiness release (ref 7) were followed.

TEST OBJECTIVES

2. The primary objective of this test was to determine ice accretion characteristics of the OV-1 engine, cowling assembly, propeller spinner, propeller, and cockpit windshield in natural icing conditions with the modified improved anti-icing system. Secondary objectives were: (1) to determine the reliability, maintainability, and failure/troubleshooting indications provided by the system and (2) to conduct engine acceleration tests with the electrical load being carried by one AC generator.

DESCRIPTION

3. The test aircraft, OV-1D(C) SN 68-15934, is a two-place, twin turboprop aircraft with a midwing design, triple vertical stabilizers and tricycle landing gear. The louvered, scarfed, shroud suppressers were not installed. A more complete description of the OV-1D is contained in the operator's manual (ref 8). The test aircraft was modified with an improved anti-ice system using three-phase power provided by new 24 KVA AC generators. A more complete description of the OV-1 modified improved anti-ice system is contained in appendix B.

TEST SCOPE

4. The OV-1 modified improved anti-icing system was evaluated in natural icing conditions from 6 to 23 March 1989 to determine ice accretion characteristics, the system's ability to handle moderate icing conditions, and system reliability. Test conditions are presented in table 1. Six flights were conducted with 6.4 productive hours of immersion in

natural icing conditions. Engine acceleration tests were conducted in 5,000 ft increments from 5,000 ft pressure altitude (Hp) to 25,000 ft Hp in clear air.

TEST METHODOLOGY

5. The JU-21 scout/chase aircraft located the required icing conditions prior to takeoff of the OV-1. The OV-1 was flown in the icing environment in a clean configuration at cruise power (60 percent torque and 1450 propeller rpm) which resulted in airspeeds from 120 to 185 knots true airspeed. The JU-21 documented the test conditions and provided air-to-air photo documentation after each icing encounter. Natural icing test techniques are further discussed in appendix C.

Table 1. Natural Icing Test Conditions

Flight Number	Average Static Outside Air Temperature (Deg C)	Average Liquid Water Content (gm/m^3)	Average Median Volumetric Diameter (microns)	Time in Cloud (hour)	Average Pressure Altitude (ft)	Equivalent Icing Condition
2	-12.0	.17	17	1.2	11,250	Light
3	-9.5	.08	N/A	1.0	12,000	Trace
4	-2.0	.10	N/A	1.1	7,000	Trace
5	-8.5	.30	14	1.0	6,970	Light
6	-8.0	.60	14	1.1	4,300	Moderate
9	-10.5	.05	N/A	1.0	11,500	Trace

RESULTS AND DISCUSSION

GENERAL

6. The OV-1 modified improved anti-ice system evaluation was conducted in natural icing at the conditions shown in table 1. Documentation is presented in figures D-1 through D-11, appendix D. Six flights (6.4 productive flight hours) in trace, light, and moderate icing conditions were completed. Additionally, 5.4 hours were flown in clear air to determine engine acceleration characteristics.

ENGINE INLET

7. The ice accretion characteristics of the engine inlet at temperatures colder than -15 degrees C were reported as a deficiency during AEFA Project No. 87-25. The actual temperature at which ice starts to accrete in the inlet was determined during this test to be -11 degrees C (figs. D-1 and D-2). The failure of the anti-icing system to keep the engine inlet clear at temperatures colder than -10 degrees C is a deficiency. The engine inlet anti-ice system should be redesigned to allow safe operation in icing conditions at temperatures colder than -10 degrees C. The OV-1 with the improved anti-icing system should be cleared into icing conditions up to and including moderate with ambient temperatures no colder than -10°C.

PROPELLER BLADES AND PLATEAUS

8. The propeller blades and plateaus are protected by a deice system as described in appendix B. The propellers accreted ice to approximately three-quarters of an inch in light and moderate icing conditions. Ice accreted on the plateaus to approximately one-half inch radially out from the plateaus. When these components deiced, airframe vibration increased and chunks of ice impacted the side of the fuselage and drop tanks causing slight damage (fig. D-3 and D-4). None of this ice was ingested by the engine. The ice accretion characteristics of the propellers with the improved deice system result in less accretion and engine ingestion hazard than the system presently used on operational aircraft. The propeller shedding characteristics are a shortcoming because of the resulting fuselage and drop tank damage.

PROPELLER NOSE AND SPINNER

9. The propeller spinner nose was anti-iced at the aluminum cap only (fig. D-5). The spinner accreted ice in a donut or crown shape. Shedding ice was observed to go down through the propeller and away from the engine inlet area. No ice damage was noted on the propeller. As reported in AEFA Project No. 87-25, splinter ice formations started growing on the spinner portion just aft of the "crown" and accumulated to a depth of 1/2 to 1 inch (fig. D-5) in light and moderate icing conditions. Grumman analysis showed that any accretions on the spinner aft of the aluminum cap would be thrown out of the engine inlet area by centrifugal force and, therefore, was not considered to be an ingestion hazard to the engine. No ice from the propeller nose and spinner was observed being ingested by the engine. The ice accretion and shedding characteristics of the propeller nose and spinner are satisfactory.

PROPELLER SPINNER AFTERBODY

10. Splinter shaped ice accreted on the spinner afterbody on the bottom portion from the 4 to 8 o'clock positions from the leading edge of the afterbody to approximately 6 inches aft (fig. D-6). During icing encounters, ice accreted at the leading edge of the afterbody and grew forward until it was dislodged by vibration or air pressure, or impacted by the rotating propeller blade shank. When a section of the accreted ice broke off, it was ingested into the inlet and, therefore, is a potential ingestion hazard to the engine, although there was no engine damage noted. At temperatures colder than -10 degrees C the splinter shaped ice accretions grew to approximately two inches by three-fourths inch in thickness prior to shedding. The ice accretion and shedding characteristics of the propeller spinner afterbody which may result in ice foreign object damage (FOD) to the engine in light and moderate icing conditions at temperatures colder than -10 degrees C are a deficiency.

WINDSHIELD ANTI-ICE

11. The numerous dropouts of one or both windshield anti-ice systems experienced during AEFA Project No. 87-25 were greatly reduced by electrical shielding. There is no system failure warning. However, four dual windshield dropouts were observed using test instrumentation during this test program. Windshield anti-ice operation appeared to contribute to the number of converter dropouts experienced (para 18). Windshield anti-ice operation also appeared to cause fluctuations of 10 to 15 degrees in the magnetic compass, contributing to the difficulty of flying the aircraft under partial panel conditions. The lack of windshield anti-ice system failure warning is a shortcoming. There was a small segment on the outboard portion of each windshield that was not heated and did accrete a small amount of ice, however, this accretion caused no significant problem other than contributing additional blind spots. The windshield anti-ice system, when operational, is satisfactory and enhances safe mission accomplishment.

PERFORMANCE AND FLYING QUALITIES

12. During AEFA Project No. 87-25 and during this evaluation performance degradations were noted for each icing condition evaluated. Moderate icing conditions were entered at 185 knots indicated airspeed (KIAS) at a cruise power setting of 1450 rpm and 60% torque. After 15 to 30 minutes in a moderate icing environment, airspeed decreased to 120 KIAS with a power setting of 80% torque. Airframe buffet was at times experienced at airspeeds as high as 150 KIAS. The deice boots were activated after approximately 1/2 inch of ice had accreted on the wing leading edge, but the boots were ineffective in removing all of the ice from the leading edges of the wings and vertical and horizontal stabilizers (figs. D-1 and D-7). At temperatures colder than -10 degrees C, the ice adhered more tightly and boot inflation was less effective in removing leading edge accretions. Additionally, large amounts of ice accreted on the inboard wing sections where no deice boots are installed (fig. D-1, D-7, and D-8). The failure of the pneumatic deicing system to remove wing and empennage leading edge ice accumulations in a moderate icing environment at temperatures colder than -10 degrees C precludes single engine level flight capability and is a deficiency. The following **WARNING** should be placed in the operator's manual:

WARNING

When flying in icing conditions, if the indicated airspeed decreases 15 knots within a 5 minute period or decreases to 145 knots with a power setting for maximum range airspeed, the airframe ice protection system may become ineffective and the icing conditions should be exited immediately.

13. The discussion in the operator's manual on pneumatic deice system operation is inadequate. The operator's manual states that the pilot should not activate the deicing system until at least one-half inch of ice has accumulated. The operator's manual does not, however, say to turn the system back off until another one-half inch of ice has accumulated. The operation instructions imply that the system can be turned on and left on, whereas it should by turned off after each cycle so that the ice does not balloon out around the boots. The pneumatic deice system for the OV-1 should allow for a single inflation cycle with a single switch movement after each appropriate ice build-up. For the current OV-1 aircraft pneumatic deice system, the operator's manual should be changed to include the following sentence in paragraph 8-6b: ... insure complete deicing of the wings and empennage. "The pneumatic deice system should then be turned off until wing leading edge ice has again accumulated to approximately one-half inch."

ENGINE DAMAGE

14. The engine was visually inspected after each flight. There was no evidence of engine damage.

AIRFRAME DAMAGE

15. Propeller and wing ice sheds occurred during deice cycles in all of the icing conditions tested. Ice departing the blades and or wings frequently hit the fuselage, empennage or fuel drop tanks. Figures D-3, D-4, and D-9 shows fuselage damage resulting from propeller ice sheds. Dents as large as 3/8 inch deep and 4 inches in diameter were observed during AEFA Project 87-25. **The propeller shedding characteristics are a shortcoming because of the resulting fuselage, empennage and drop-tank damage.**

PROPELLER DAMAGE

16. During AEFA Project No. 87-25 and during this evaluation there was propeller damage caused by ice departing from the windshield wipers and going through the propellers. Windshield wiper ice accumulation is shown in figure D-10. The ice accretion and shedding characteristics of the windshield wipers are a shortcoming due to the damage caused to the propellers.

AC GENERATORS

17. There were no AC generator dropouts and no generator shaft breakage during this evaluation. Grumman Aircraft Corporation provided increased strength shafts (3000 versus 1200 inch-pounds previously) and changed the frequency of on-off cycling of the anti-ice controllers. The AC generator drop off speed was reduced to 8200 rpm (ground idle). This eliminated uncommanded AC generator dropouts. The operational characteristics of the AC generators are satisfactory.

CONVERTER OPERATION

18. The converters dropped off line frequently. The number one converter failed during the initial engine acceleration tests. A Test Incident Report (TIR) was submitted and a copy is included in appendix F. Numerous converter dropouts were experienced during windshield anti-ice operation. There were no converter dropouts when the windshield anti-ice system was not being operated. Each time a dropout occurred, the Master Caution light with its associated annunciator light illuminated and a manual reset of the converter was required. The numerous converter dropouts during normal operations, the increased pilot workload required to reset them, and the resulting pilot distractions caused by illumination of the Master Caution light are a hazard during flight in instrument meteorological conditions, will preclude successful completion of operational missions, and are a deficiency.

INVERTER OPERATION

19. The numerous inverter dropouts experienced during AEFA Project No. 87-25 were no longer a problem, but the ensuing delay and actions required to re-establish flight essential (attitude indicator and radio magnetic indictor) and normal inverter loads should a failure occur remain a deficiency. If the inverter drops off-line or if the pilot needs to go to the BACKUP position to power the converter bus, he must wait several seconds with the inverter switch in the OFF position before the switch can be reset back to NORMAL or to BACKUP. This is an extreme nuisance during flight in instrument conditions. There is another approximately 10 second delay prior to restoration of AC instrument power after the switch is positioned. The ensuing delay and actions required to re-establish flight essential and normal inverter loads following inverter dropouts are a hazard during flight in instrument meteorological conditions, will preclude successful completion of operational missions and are a deficiency.

ENGINE ANTI-ICE SWITCH

20. The engine anti-ice switch labeling has reverse sensing as discussed in AEFA Project No. 87-25. The reverse sensing and mislabeled engine anti-ice switch (all other switches have their normal position up, the anti-ice switch has its normal position down) remains a shortcoming. The EMER position should be labeled "BACK-UP' due to the implication that there is an emergency if the switch is in that position.

ICE DETECTOR

21. When the ice detector senses ice it activates the anti-ice/deice system. The system then starts through an Operational Readiness Test (ORT) with resultant illumination of the Master Caution Light and the #1 and #2 ANTI-ICE segment light. The lights go out in 10 seconds if the system passes the ORT and turns the system ON. However, the illumination of the Master Caution light during anti-ice system activation is a nuisance to the pilot and is a shortcoming. Activation of the anti-ice system by the ice detector should be an advisory only. During this evaluation, the ice detector would intermittently not activate the system. The switch was placed to the emergency position and the system operated normally. If the switch was placed back to normal, the system would again turn itself on. One Rosemount probe was changed, however the same situation occurred again. The reason for the intermittent failure of the system to turn itself on could not be determined. The intermittent failure of the ice detector to activate the anti-ice/deice system is a deficiency. A TIR was submitted and a copy is included in appendix F.

ANTI-ICE TEMPERATURE CONTROLLER

22. The problem with the anti-ice controller not functioning properly after passing its ORT appears to have been an idiosyncrasy of the electrical problems experienced during AEFA Project No. 87-25. The same controller was again installed and used during this evaluation with no further problem.

PITOT TUBE

23. During this evaluation in light and moderate icing conditions, ice accretion started on the vertical base of the pitot tube and built forward and up around the pitot tube (figs. D-11 and D-12), never touching the horizontal portion of the pitot tube itself. Airspeed indications were disrupted (the airspeed reading was from 50 to 90 KIAS), after the ice accreted to within one-half inch of the leading edge of the pitot tube opening. The pitot heating element was checked in accordance with maintenance procedures and was verified to be working as required. The aircraft manufacturer has also verified that the pitot heating element was producing specification heat output. The ice accretion characteristics of the pitot tube which result in erroneous airspeed indications are a deficiency.

ENGINE ACCELERATION TESTS

24. Engine acceleration tests were conducted in 5000 foot increments from 5000 feet pressure altitude to 25,000 feet pressure altitude, at the conditions shown in tables E-1 through E-5. Tests were conducted with each engine individually, with bleed air on and with bleed air off, with each engine carrying its own electrical load and with each engine carrying both engines electrical load to determine if stalls would occur or limits would be exceeded. It should be noted that no mission equipment was being powered during the engine acceleration tests except the anti-ice/deice equipment. The accelerations were accomplished by stabilizing at the desired N1 speed, insuring that the appropriate electrical

load was established, then applying full throttle within one-half second, simultaneously starting a stopwatch, then stopping the stopwatch when N1 speed peaked. Engine acceleration data is presented in tables E-1 through E-5. There were no compressor stalls and no limits were exceeded. N1 acceleration times were notably increased for one engine carrying the full electrical load with the greatest acceleration time from 70 percent N1 to maximum N1 being 17 seconds, compared with 11 seconds for the engine carrying only its own electrical load, at 25,000 feet. The engine acceleration characteristics of the OV-1 aircraft with one engine carrying the full anti-ice/deice electrical load are satisfactory.

CONCLUSIONS

GENERAL

25. Seven deficiencies and five shortcomings were identified during the OV-1 Modified Improved Anti-Icing System evaluation. The inlet area ice accretion, though still presenting potential engine ice ingestion damage is much less than accretion on the standard OV-1 inlet. The engine acceleration characteristics with one engine carrying the full anti-ice/deice electrical load are satisfactory. Specific conclusions which were previously reported during AEFA Project No. 87-25 are denoted with an asterisk(*).

ENHANCING CHARACTERISTICS

*26. The windshield anti-ice system, when operational, enhances safe mission accomplishment (para 11).

DEFICIENCIES

27. The following deficiencies are listed in order of importance:

*a. The failure of the pneumatic deicing system to remove wing and empennage leading edge ice accumulations in a moderate icing environment at temperatures colder than -10 degrees C precludes single engine level flight capability and is a deficiency (para 12).

*b. The failure of the anti-icing system to keep the engine inlet clear at temperatures colder than -10 degrees C (para 7).

*c. The ice accretion and shedding characteristics of the propeller spinner afterbody which may result in ice FOD to the engine in light and moderate icing conditions at temperatures colder than -10 degrees C (para 10).

d. The ice accretion characteristics of the pitot tube which result in erroneous airspeed indications (para 23).

e. The intermittent failure of the ice detector to activate the anti-ice/deice system (para 21).

*f. The numerous converter dropouts during normal operations, the increased pilot workload required to reset them, and the resulting pilot distractions caused by illumination of the Master Caution Light (para 18).

*g. The ensuing delay and actions required to re-establish flight essential (attitude indicator and radio magnetic indicator), and normal inverter loads following inverter dropout (para 19).

SHORTCOMINGS

28. The following shortcomings are listed in order of importance:

*a. The propeller shedding characteristics are a shortcoming because of the resulting fuselage, empennage and drop tank damage (paras 8 and 15).

b. The ice accretion and shedding characteristics of the windshield wipers due to the damage caused to the propellers (para 16).

*c. The lack of windshield anti-ice system failure warning (para 11).

*d. The illumination of the Master Caution Light during anti-ice system activation (para 21).

*e. The reverse sensing and mislabeled engine anti-ice switch (all other switches have their "normal" position up, the anti-ice switch has its normal position down) (para 20).

RECOMMENDATIONS

29. Correct the deficiencies listed in paragraphs 27a through 27c prior to release of the aircraft for flight in icing conditions at temperature colder than -10°C.

30. Correct the deficiencies listed in paragraphs 27d and 27e prior to release of the aircraft for flight in icing conditions at any temperature.

31. Correct the deficiencies listed in paragraphs 27f and 27g prior to release of the aircraft for operational mission or for flight in instrument meteorological conditions.

32. Correct the shortcomings listed in paragraph 28 as soon as practicable.

33. The following **WARNING** should be placed in the operator's manual.

WARNING

> When flying in icing conditions, if the indicated airspeed decreases 15 knots within a 5 minute period or decreases to 145 knots with a power setting for maximum range airspeed, the airframe ice protection system may become ineffective and the icing conditions should be exited immediately (para 12).

34. Modify the pneumatic deice system for the OV-1 to allow for a single inflation cycle with a single switch movement after each appropriate ice buildup (para 13).

35. The operator's manual for the current OV-1 pneumatic deice system should be changed to include the following sentence in paragraph 8-6b: ...insure complete deicing of the wings and empennage . "The pneumatic deice system should then be turned off until wing leading edge ice has again accumulated to approximately one-half inch." (para 13).

36. The EMER position should be labeled "BACK-UP" due to the implication that there is an emergency if the switch is in that position (para 20).

37. Activation of the anti-ice system by the ice detector should be an advisory only (para 21).

APPENDIX A. REFERENCES

1. Letter, AEFA, DAVTE-TB, July 1981, subject: Report, Limited Artificial Icing Tests of the OV-1D.

2. Final Report, AEFA Project No. 81-21, *Limited Artificial and Natural Icing Tests of the OV-1D (Re-evaluation)*, June 1982.

3. Final Report, AEFA Project No. 87-25, *Evaluation of the Improved OV-1D Anti-Icing System*, April 1988.

4. Letter , AVSCOM, AMSAV-8, 13 December 1988, subject: Evaluation of the Improved OV-1D Anti-icing System, Phase II, Project No. 87-25-1. (Test Request)

5. Test Plan, AEFA Project No. 87-25, *Evaluation of the Improved OV-1 Anti-Icing System*, August 1987.

6. Technical Manual, TM 55-1510-213-10, *Improved Electrical Anti-Icing and AC Electrical System for A/C 67-15934, OV-1D(C), Mohawk Aircraft, Prototype Operator' Manual*, 4 August 1978.

7. Letter, AVSCOM, AMSAV-E, 8 February 1989 subject: Airworthiness Release for OV-1D(C), S/N 68-15934, for AEFA Project No. 87-25-1, Evaluation of the OV-1D Improved Anti-Icing System, Phase II.

8. Technical Manual, TM 55-1510-213-10, *Operator's Manual OV/RV-1D Aircraft*, 4 August 1978.

9. Final Report, AEFA Project NO. 83-10, *Verification of U-21A Cloud Parameter Measurement Equipment and Comparison of Natural and Artificial Ice Accretion Characteristics on Rotor Blade Airfoil Sections*, to be published.

APPENDIX B. DESCRIPTION

GENERAL

1. A description of the ice protection system currently installed on operational aircraft is contained in reference 9, appendix A. For the U.S. Army Aviation Engineering Flight Activity AEFA Project 87-25 evaluation, aircraft OV-1D(C) serial number 68-15934, was modified with an improved Anti-Ice System using three-phase AC electric power provided by two new 24 Kilo-Volt-Ampere (KVA) AC generators. The system description is repeated here as it was for AEFA Project No. 87-25. The changes made for this evaluation are annotated.

AC ELECTRICAL POWER

2. The AC power system was modified by a preliminary engineering change proposal GR-OV-334 and consists of the following changes:

OLD SYSTEM: (Installed on Operational Aircraft)
(2) 6.5 KVA 115V variable frequency AC electrical generator system
(1) 750 KVA rotary inverter
(1) 2500 VA rotary inverter
NEW SYSTEM: (Improved Anti-Ice System, Installed for AEFA Project 87-25 and for this evaluation)
(2) 24 KVA 115V 400 Hz nominal variable frequency 306 to 480 Hz generator
(2) 10 KVA 115V 400 Hz converter
(1) 10 KVA 115V 400Hz static inverter

A schematic of the new AC electrical system is shown in figure B-1.

3. The AC system contains two AC buses: an inverter bus and a converter bus. The inverter/converter bus crossover relay (K77) automatically combines the two buses through the weight on wheels switch. This is required because the converter output is be cut off when its respective AC generator speed is reduced below 8520 rpm. This was changed from 9200 rpm for project No. 87-25. The 10KVA inverter will supply mission loads to the converter bus through the inverter/converter bus crossover relay to provide preflight operations to the mission equipment. Failure of the inverter for any reason, such as an under-voltage condition not corrected within 4 seconds, causes the inverter output relay (K76) to de-energize. The inverter input relay (K11) de-energizes, taking the inverter off line. Whenever the output relay (K76) de-energizes, it will automatically energize the inverter/converter bus crossover relay (K77) through normally closed contacts.

4. The pilot can manually energize the crossover relay to provide inverter power to the converter bus by moving the inverter selector switch to the backup mode. If the converters are lost, the inverter will provide power to the windshield anti-ice system as well as the flight essential loads through the inverter/converter crossover relay contacts.

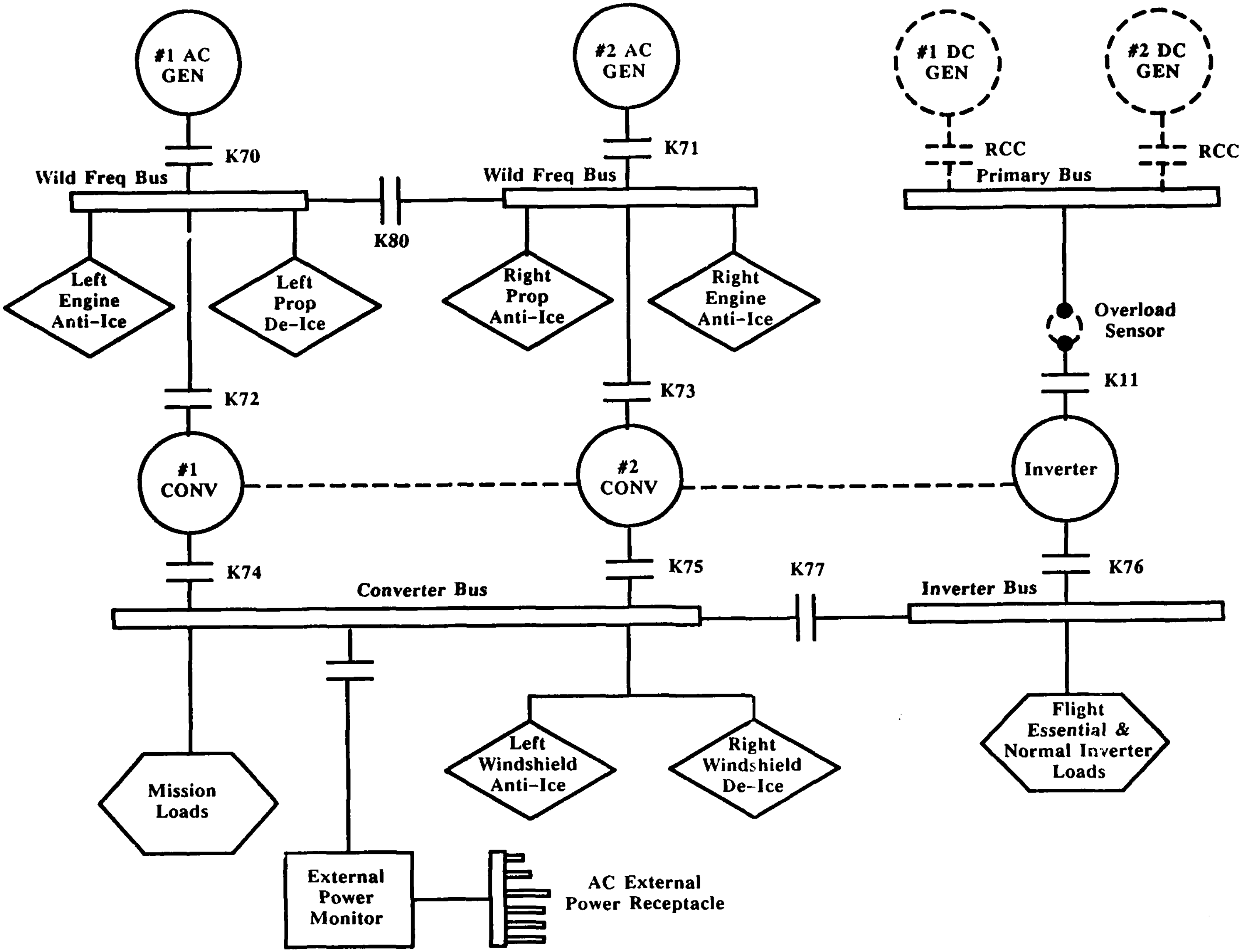

Figure B-1. AC Electrical System Schematic

AC GENERATORS

5. The AC generators are manufactured by Leland Electrosystems, Inc. and each provide 24 KVA, three-phase power of 115/200 volts, variable frequency (400 Hz nominal) to its respective left or right 10 KVA converter at 11200 rpm. At lower speeds, the generator output is gradually reduced to 12.5 KVA at 8900 rpm. Below 8200 rpm (changed from 8900rpm for AEFA Project No. 87-25), the AC generators will cut off. During normal operations, the No. 1 AC generator supplies the left side of the anti-ice /deice system and also the No. 1 converter. The No. 2 AC generator supplies the right side of the anti-ice/deice system and the No. 2 converter. In the event of a loss of either generator, a warning light will illuminate in the cockpit indicating failure.

6. The pilot can manually activate an AC generator crossover bus relay (K80) which permits one AC generator to power the total anti-ice/deice system and both converters. The AC generator crossover bus relay is normally open. Upon failure of either generator in icing conditions, the pilot would dump the mission loads and energize the AC generator crossover bus relay (K80), allowing one generator to feed both converters which would supply windshield anti-ice. Further, this switching action prevents the possibility of paralleling the generators together which would result in an over-current condition.

7. The following warning light indicators are available within the cockpit for the pilot:

#1 AC GEN
#2 AC GEN
#1 CONV
#2 CONV
INV
#1 ANTI-ICE
#2 ANTI-ICE

8. Each AC generator is protected by its respective Generator Control Unit (GCU), which is located in the lower nacelle area. The GCU senses over/under-voltage, under-speed, short circuit, over-current, over-temperature, and feeder faults. An under-speed fault will occur whenever the generator speed falls below 8750 rpm and resets automatically when rpm increases above 8900 rpm. Manual reset is accomplished by positioning the appropriate AC GEN switch to OFF and then back to ON. The system will only reset if the fault is cleared.

10 KVA CONVERTERS

9. Two 10 KVA converters, manufactured by Leland Electrosystems, Inc., have been provided to convert 115/200V, wild frequency (nominal 400 Hz), from the AC generators and convert it to a regulated 115/200V AC precise 400 Hz output. The two 10 KVA converters normally operate in parallel to a common bus and provide power to the camera system, side looking airborne radar set or the infrared set (IR) and the windshield anti-ice system. The converters are located in the top of the equipment bays above the KD-76 camera, fuselage station (FS) 204, and the aft KA-60 camera, FS 248. Each converter weighs approximately 90 lb.

10. The converter assembly is divided into two major power stages; AC-DC Stage, and DC-AC Stage. These stages, when combined, produce regulated 400 Hz power.

First Stage: AC-DC Stage takes 115/200 VAC wild frequency from the AC generator, and rectifies and filters it to produce an un-regulated ±135 VOLTS DC.

Second Stage: Converts ±135 Volts DC into a regulated 115/200V AC precise 400 Hz using Sinusoidal Pulse-Width Modulation Technique. This technique switches the ±135V DC into positive and negative half cycles of 400 Hz by driving the output with sine-weighted pulses.

11. A microcontroller is constantly monitoring the converter input/output and also synchronization/load sharing conditions. It will respond to the following failure conditions by actuating a fault flag and removing the converter from on-line: Input over/undervoltage; output over/undervoltage; output over/under frequency; overtemperatrue; short circuit; wave form distortion; input under frequency.

12. The converters have a rated output of 10 KVA continuous, or 15 KVA overload for five seconds and 12.5 KVA overload for five minutes with an efficiency of approximately 80%. The converters are controlled by the AC CONV No.1 and No.2 switches on the left pilot's overhead panel. The ON position enables the converter to perform normal regulatory and monitoring functions. The OFF position disables the converter and establishes a reset. The converter requires that its respective AC generator be on-line (unless K-80 relay is closed) and above 8520 rpm. There are no conditions in which the converter will automatically reset after a fault has occurred.

13. The failure of one 10 KVA converter does not present a problem. The remaining converter can supply the entire mission and windshield anti-ice load. Failure of both converters will cause mission loads to be lost, however, the 10 KVA inverter can power mission equipment and windshield anti-ice for this condition by energizing the inverter/converter bus crossover relay (K77).

STATIC INVERTER

14. The AC power generation system utilizes one static inverter which is manufactured by Leland and supplies power to operate flight essential instrumentation and normal inverter loads. The inverter's function is to take input 28 volts DC from the primary bus and generate 115/200V AC. It is capable of operating individually or in conjunction with two converters to supply a common three phase AC bus. The inverter and converter have identical microcomputer-controlled DC-AC power inverter stages and are designed to precisely load share, however, the inverter has a ten second time delay built-in before output power is provided.

15. The normal rated output of the inverter is 10 KVA. The inverter has built-in protection for the following fault conditions: overtemperature; wave form distortion; over/under frequency condition; over/under input and output voltage condition. It is also protected by an overload sensing control in the 28 VDC primary bus input. Excess input current through the overload sensing control results in shutdown of the inverter with

illumination of the INV annunciator and master caution light. All of the above failures require a manual reset by positioning the INV switch to OFF and then back to NORMAL. The system will only reset if the fault has been cleared. If the inverter fails with the INV switch in the NORMAL position, the converter bus power is automatically connected to the the inverter bus through K-77 relay. The inverter bus will be powered by the converter bus if the inverter fails in the NORMAL or BACKUP mode. If both converters fail the inverter switch needs to be placed to the OFF position for 5 seconds, then to BACKUP. The inverter will provide sufficient AC power to operate normal aircraft systems and windshield anti-ice. If the inverter fails while both converters are failed, no backup AC source and no windshield anti-ice/deice is available.

AC EXTERNAL POWER

16. The AC system includes provisions for connecting an external source of regulated, three-phase, 115/200 V, 400-cycle AC power to the converter bus. The AC external power receptacle is located on the left side of the equipment compartment No.4. The AC receptacle has provisions for protection from overvoltage, undervoltage, overfrequency and reverse phase rotation. Therefore, any anti-ice/deice system function requiring wild frequency AC power will not receive ground power through the AC external receptacle but the windshield anti-ice and mission loads can receive ground external power.

ANTI-ICE/DEICING SYSTEM DESCRIPTION

General

17. Grumman Aircraft Corporation designed and installed an improved electrical anti-ice system (tested on Project AEFA 87-25 and for this evaluation). The existing method of ice prevention of the cowl inlet and spinner on the Mohawk OV-1D utilized two 6.5 KVA AC deicing generators. The heating elements are energized by single phase 115V AC. The new anti-ice/deice system contains two 24 KVA AC generators which anti-ice the cowl inlet and spinner. The pneumatic deice system was not altered. The design used NASA Lewis' environmental wind tunnel test results to establish baseline data. New wiring provisions were installed and the pilot's overhead panel was redesigned to include new anti-ice switching. An analytic propeller performance evaluation was conducted by propulsion engineering. Newly designed power takeoff gears were provided by Western Gear Ind. and installed by Corpus Christi Army Depot personnel. The Improved Electrical Anti-Icing system is comprised of two segments on each side of the aircraft: (1)engine/propeller anti-ice/deice and (2) windshield anti-ice.

Spinner Anti-ice/Deice System

18. Engine anti-ice/deice consists of heating elements installed on the spinner highlight, spinner plateaus, and propeller blades, all of which have a protective metal covering, plus fixed and removable engine nacelle cowls, cowl split-lines, and cowl struts. A detailed design of the cowl and strut heater elements is shown in figure B-2. The No. 1 AC generator normally supplies three-phase wild frequency power to the left anti-ice/deice

Area	Wrap Distance From Highlight - (Inches)	Power Density - W/In^2
1.	-2.0 to -1.5	8.5
2.	-1.5 to -0.5	11.0
3.	-0.5 t0 0.5	12.5
4.	0.5 to 1.5	11.0
5.	1.5 to 3.5	9.0
6.	3.5 to 4.5	8.0
7.	4.5 to 9.0	7.0
8.	9.0 to 10.0	6.0
9.	10.0 to 12.0	5.0
10.	12.0 to 13.75	3.5

Cowl Power - 7500W (nominal)
Struts and Splitline - 700W
Total Cowl Anti-Icing System Power - 8200W (nominal)

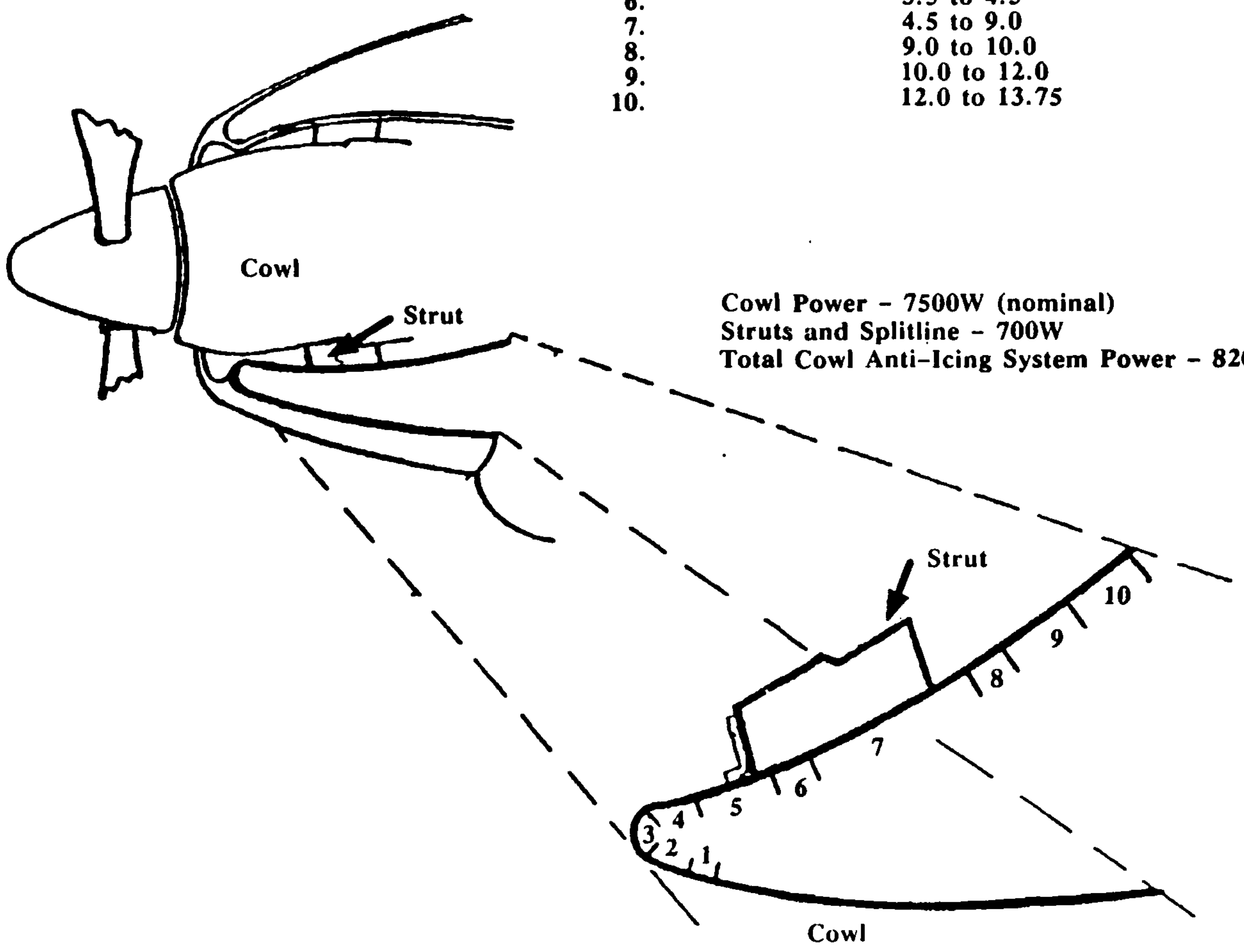

Figure B-2. Location and Power Densities of the Cowl and Strut Heater Elements

resistive load and No. 2 AC generator supplies the right side. Each side of the anti-ice/deice system block diagram is shown in figure B-3.

19. The anti-ice/deice system contains the following caution/warning lights located on the caution annunciator panel.

a. "#1 ANTI-ICE"
b. "#2 ANTI-ICE"
c. "ANTI-ICE ON"

20. The anti-ice caution/warning lights illuminate to indicate a failure of the No. 1 and/or No. 2 anti-ice/deice system. The "ANTI-ICE ON " is an advisory light which indicates that the system is activated. This advisory light illuminates only on the ground, with the aircraft's weight on wheels and the anti-ice/deice system "ON".

21. The anti-ice/deice system may be energized manually by turning the anti-ice switch to either the "NORM" (normal) or "EMER" (emergency) position. The engine anti-ice switch is located on the left overhead panel. The "EMER" position of operation directly controls 28V DC power to the anti-ice/deice system, bypassing the ice detector and activating the spinner, propeller, and engine ice protection systems. In the "NORM" position, the anti-ice position, the anti-ice switch supplies 28V DC to the ice detector which operates in conjunction with an anti-ice relay. The ice detector consists of a probe which senses the presence of ice and provides an output to activate the aircraft's anti-ice/deice equipment only in the "normal" mode of operation.

22. The Rosemount model DT-5041-A ice detector is comprised of two oscillators. The magnetostrictive oscillator (probe) without icing conditions oscillates in the range of 40.2 KHz. A mixer compares the difference in the frequency of both oscillators. As the ice build-up on the probe increases, a decrease occurs in the operation frequency of the mixer. If the delta frequency is between 5 and 350 Hz, the ice detector will energize an anti-ice relay which supplies 28V DC power to the anti-ice/deice controllers. The output remains on for 60 seconds after icing conditions are sensed. A second voltage remains on for 5 seconds after ice is sensed which activates an internal heater that deices the probe.

23. The anti-ice system contains two microprocessor temperature controllers. The main functions of the microprocessor temperature controller are: temperature sensing, power switching, self-test, fault detection, and signaling. The temperature controllers control AC power to the spinner and cowl anti-ice heating elements to maintain an ice free engine inlet. Both temperature controllers can operate continuously or intermittently without any adjustments required from the pilot.

24. The temperature controller interfaces with two temperature sensors per nacelle. One temperature sensor is located in the fixed section of the cowl, the other in the removable cowl section. The temperature controller possesses the capability, such that, should one sensor fail the remaining sensor controls the anti-ice system. The pilot anti-ice caution/warning light will illuminate for an open or shorted sensor. If both sensors fail, the temperature controller fails "safe", that is, power is maintained to the heater elements. In flight without icing conditions, the temperature generated by the cowl heater elements cannot cause structural damage to the engine cowl. A correlating coefficient between the

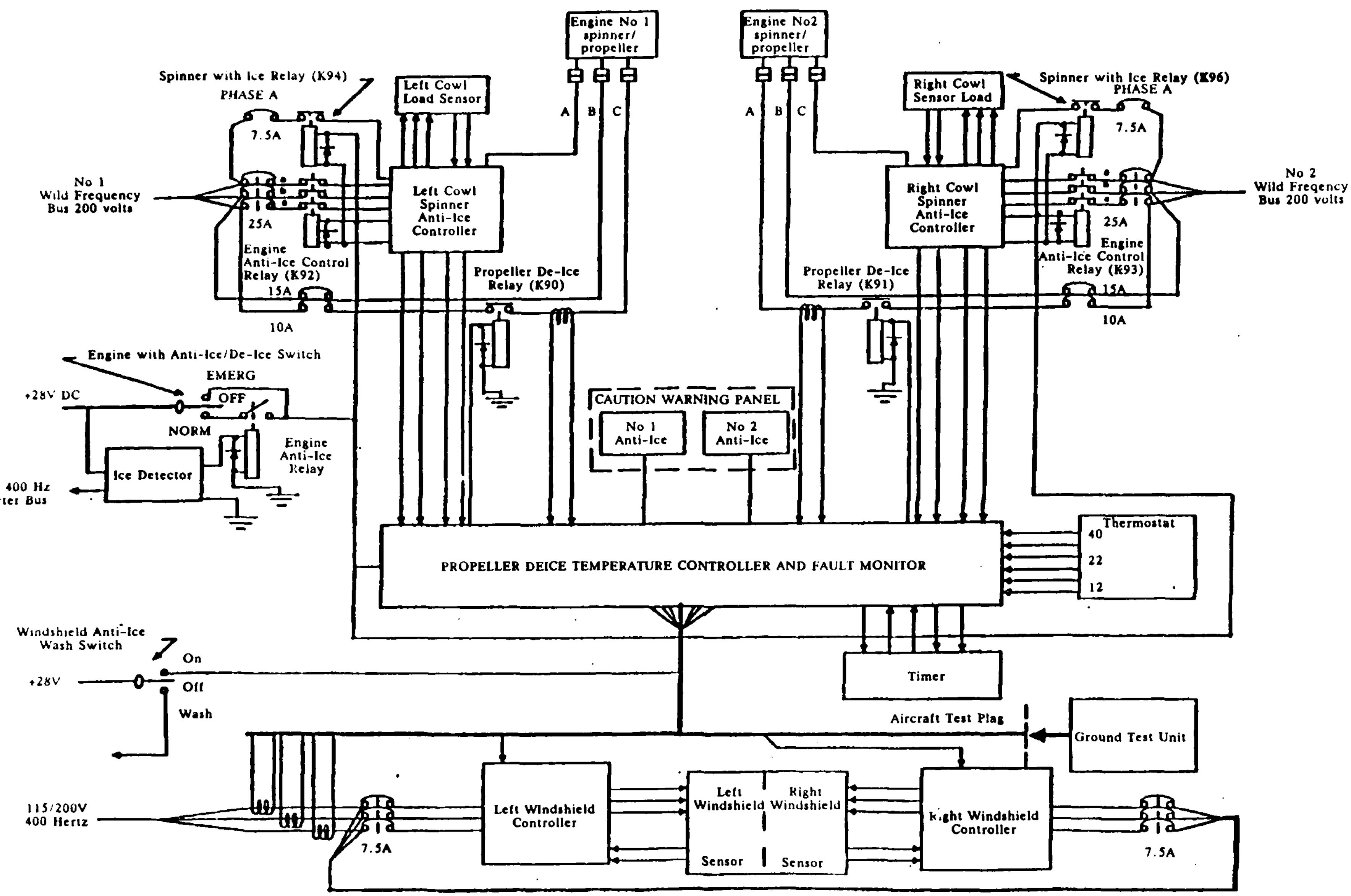

Figure B-3. Anti-Ice System Block Diagram

power demand of the cowl and spinner is stored in an erasable programmable read only memory. The spinner anti-ice heating elements can, therefore, be controlled from the cowl thermal sensors through a correlation of the cowl temperature control signal. The microprocessor thus develops a power control signal for the single phase power switch of the controller which supplies power to the spinner heater. No temperature sensors are located within the spinner. The spinner heating element remaining energized without icing will cause blistering of the spinner until the heater elements burnout. The anti-ice system is lost to engine No.1 spinner highlight but the engine anti-ice remains operational.

25. The spinner-cowl temperature controller contains a built-in test capability which includes an on-board operational readiness test (ORT) and failure detection/isolation capability. The ORT checks out all phases of the spinner-cowl temperature controller and determines the operational readiness of the heating system. The ORT is activated for less than 10 seconds of elapsed time from power up, and then returns to its normal operating mode. During this test, 11 fault signals are inhibited and the anti-ice caution/warning light is illuminated. If the anti-ice warning light fails to extinguish after an ORT test, a fault exists somewhere within the anti-ice system. During the ORT test, the two heater elements are tested sequentially at moderated duty cycles to maintain average system power below 25% of maximum power. The ORT test is capable of being performed on the ground or during flight. The pilot can manually initiate the ORT with a switch located within the cockpit. This switch is in the ice protection circuit breaker panel (sloping console).

26. Fault detection and isolation are implemented by the microprocessor through software checks and special sensors and circuitry to monitor the various voltage and currents at critical points in the heater power flow. The following system faults can be detected:

a. input power loss
b. input voltage dropout or imbalance exceeding 10%
c. open or shorted power switch
d. open or shorted heater elements
e. open or shorted sensors
f. under/overtemperature

27. The microprocessor temperature controller contains a timer which automatically resets the controller if it becomes lost in a program as a result of an isolated transient or a momentary power loss. The processors and the critical circuit components are buffered and isolated from the sensors, heater firing circuits, and other signals. This prevents a fault in one heater or sensor circuit from propagating and causing damage throughout the controller and also protects the processing section of the controller.

28. If a fault occurs, a fault signal is developed which illuminates the anti-ice caution/warning light. This light informs the pilot that a fault has occurred within the anti-ice/deice system. Four additional fault signals trip fault indicators (flags) in the fault monitor to identify the nature and area of failure.

29. Detected faults that present safety hazards will cause external relays that supply primary power to the system to be de-energized. The controller will remove power within

2 seconds of a fault occurrence for dropouts and imbalance levels exceeding 10%. The voltage at the output of each of the solid state passive elements is also monitored and compared to its input. A shorted or open input pass element will trip the fault monitor flag and then primary power will be removed. The current is also monitored through each phase of each channel. A shorted or open heater circuit will be detected and power to that load will be removed. Open or shorted temperature sensors and under/overtemperature conditions will be sensed by comparators which will signal the microprocessor of the fault. The controller signals the appropriate fault line and if necessary removes power from a particular load.

30. The spinner-cowl temperature controller contains a support equipment connector on its front panel to facilitate diagnosis of the anti-ice system failures. Production plans are for a computer controlled type of bench test equipment which will be plugged into this connector to perform function checkout of the system and isolate failures to the module/submodule level of the system.

FAULT MONITOR AND DEICE CONTROLLER

31. The deice system contains one fault monitor and deice temperature controller which controls the ice protection systems for both engines. The fault monitor and deice controller perform the following major functions:

a. Monitoring and fault display of the propeller deice system.

b. Display faults provided from both spinner-cowl temperature controllers.

c. Provides interface for the anti-ice spinner-cowl temperature controllers and for the deice timer and three setpoint thermostat.

d. The system provides an ORT for the propeller deice system, windshield anti-ice system, fault monitor interface circuits, and the caution panel anti-ice display when used with an anti-ice ground test set.

32. The system uses 200V AC to energize the heating elements on these surfaces. A timer establishes three different time periods to energize heater elements which are dependent on thermostat settings. Above 4.4 degrees C ambient temperature, The propeller timer and thermostat circuitry provide an inhibit of propeller deicing. A three element thermostat (mercury columns) is used which selects one of the three timer cycles for the propeller deice system. The timer cycles are : between 4.4 and -5.6 degrees C, a 5 second "on" - 60 second "off" cycle; between -5.6 and -11.1 degrees C a 10 second "on" - 60 second "off" cycle; below -11 degrees C, a 20 second "on" - 60 second "off" cycle. The output from the propeller timer is used to control the propeller fault monitor during the "on" portion of each cycle. A cockpit test connector is provided which allows ground testing to be performed on this part of the propeller deice control system. During flight, continuous fault monitoring is provided for cowls, spinner, and propellers. A current transformer is located in each of the two propeller power lines to the propeller heating element. The current transformers sense the input power and compares its signal with a reference level. If a difference in signal is detected, one of the fault indicator flags trip and the anti-ice caution/warning light will illuminate. The windshield anti-ice system can only be checked with a ground test set; no fault monitor provision exists during flight.

33. The fault monitor system identifies the following fault conditions and provides displays on the fault unit by individual fault indicators (flags):

a. loss of input primary power
b. cowl heater and sensor faults
c. spinner load faults
d. controller faults
e. deice propeller faults

34. The presence of any of these faults will trip a fault indicator flag on the unit and also illuminate the anti-ice caution/warning light on the caution annunciator panel. If the fault should clear, the warning light will extinguish automatically, however, the fault indicator will remain tripped and can only be reset by use of a ground test set. The fault monitor contains circuitry for a one second test signal for ORT of the propeller and windshield fault test.

WINDSHIELD ANTI-ICE SYSTEM

35. The existing method of deicing OV-1 operational aircraft windshields is by spraying a mixture of alcohol and water on the windshield through one restrictor. The duration of fluid during the deicing cycle is approximately 10 to 15 minutes. This windshield ice protection system does not always clear the windshield sufficiently. The new windshield anti-ice system electrically heats both the pilot's and observer's windshield. Each windshield anti-ice system load is approximately 2 KVA. The windshield heating elements consist of a layer of electrically conductive metal oxide (nesatron) with integral bus bars which are applied to the interior surface of the outboard layer glass.

36. The electrically heated windshield interfaces with a windshield heater controller and a thermal sensor. The windshield anti-icing system contains two windshield heater controllers which are both operated by one "on-off-wash" switch located on the left overhead panel. The windshield heater controller is a device which senses and controls three phase AC converter power into the windshield and attempts to maintain a 110 degree F setting. The No. 1 controller controls power to the pilot's windshield heater elements, the No. 2 controller to the observer's heater elements. The windshield controller operates in a closed loop with two temperature sensors connected in parallel. These temperature sensors are embedded in the interlayer of the windshield and have a positive temperature/resistance characteristic. The controller turns "on" when the resistance of the two sensors in parallel decreases to 168 (±1.0 ohms above the 168 ohm turn on value). If either sensor develops short circuit fault, the controller is designed to automatically open the power circuit. The controller also turns off for loss of DC Power. The windshield controller contains a fail safe provision which de-energizes the heater elements for internal controller failures (i.e., shorted controller output power switch). This feature precludes overheating the windshield which could cause delamination or structural failure.

37. Two 10 KVA converters supply regulated three phase 400 Hz power to the converters mission bus. The windshield anti-ice power is normally supplied from the

converters mission bus for AC generator shaft speeds above 8520 rpm. For loss of both converters or AC generator speeds below 8520 rpm, the windshield system can be powered from the DC sourced 10 KVA inverter by moving the inverter switch to its backup position.

38. The pilot's and observer's windshields are independently controlled by separate windshield controllers. Loss of the pilot's windshield controller, sensor or heater elements will not affect the anti-ice operation of the observer's windshield and vice-versa.

39. During flight, there is no fault indication to the pilot if either side of the windshield system is lost. The pilot can determine by observation if either side of the windshield anti-ice system is lost by ice accumulation or by sensing if the windshield is warm. After failure of one side or the entire windshield anti-ice system there exists no means of restoring the windshieid anti-ice system. If icing conditions exist, the pilot will be forced to attempt viewing through his side window to land the aircraft.

40. The windshield anti-ice system utilizes three current transformers to monitor each of the input three-phase circuits. This power input sense signal is compared to a reference level. No warning signal or fault signal is tripped if the fault occurs during flight. The windshield system can only be checked with use of a ground test set, which is plugged into a test jack located in the cockpit.

APPENDIX C. TEST TECHNIQUES AND DATA ANALYSIS METHODS

NATURAL ICING

1. A JU-21A scout/chase aircraft equipped with a cloud particle measuring system was used to locate and document the icing conditions. The U-21 was also configured with a bubble photographic window installed in the cabin and was used as a photographic platform to provide documentation of the ice accreted on the test aircraft. The scout/chase aircraft would locate the desired icing conditions and radio the location and icing conditions to the test aircraft before it entered the icing environment. The U-21 would then exit the icing conditions and loiter in the area to facilitate a rapid in-flight join-up with the test aircraft after it exited the cloud for photographic documentation. The OV-1 was flown in the icing environment in a clean configuration initially with cruise power (60% torque and 1450 propeller rpm) which produced approximately 185 knots true airspeed (KTAS). The test aircraft's anti-ice and deice equipment was used while in the icing conditions.

CLOUD SAMPLING EQUIPMENT

2. The cloud measurement package consists of the following equipment: a Particle Measuring Systems, Inc (PMS) Forward Scattering Spectrometer Probe (Model FSSP-100), a PMS optical array cloud droplet spectrometer probe (model OAP-200X), Rosemount total temperature sensor and display, Cambridge model 137 chilled mirror dew point hygrometer and display, Cloud Technology Inc. model LWH-1 (Johnson Williams type) liquid water content (LWC) indicator system, Small Intelligent Icing Data System (SIIDS), and two visual accretion devices: Aeroplane & Armament Experimental Establishment's Vernier Accretion Meter (Harvey-Smith) and a Small Airfoil Section probe (OH-6 tail rotor section). Figure C-1 and C-2 show the exterior of the aircraft with the probes in place, while figure C-3 shows the interior instrumentation rack with displays.

3. Each PMS probe projects a collimated helium-neon laser beam normal to the airflow across a small sample area. In forward flight, particles passing through the beam (sample area) are counted and measured into 15 size channels per probe, each probe operating over a different size range. While these probes are primarily intended as particle sizing devices, an LWC can be calculated from the drop size measurement and number count within the sample volume relative to airspeed.

4. The FSSP-100 determines particle size by measuring the amount of light scattered into the collection optics aperture as the particles pass through the laser beam. A pulse height analyzer compares the maximum amplitude of the scattering signal pulses with a reference voltage derived from a separate measurement of the illuminating light signal. The pulse height analyzer output is encoded to give the particle size in binary code, and resolves particle sizes from 2 to 47 μm into 15 equally spaced increments 3 μm wide. It is capable of sizing particles having velocities of 20 to 125 meters/sec (39 to 243 knots.) A gate output signal provides a measure of particle transit time, and a velocity averaging counter and control system determines an average transit time. The system automatically rejects particles with transit times less than average since these are susceptible to edge effect errors which result from particles passing through regions of less than maximum intensity. A laser beam width of 0.186 mm and depth of field of 2.76 mm provides a total sample area of 0.513 mm^2 (before velocity reject).

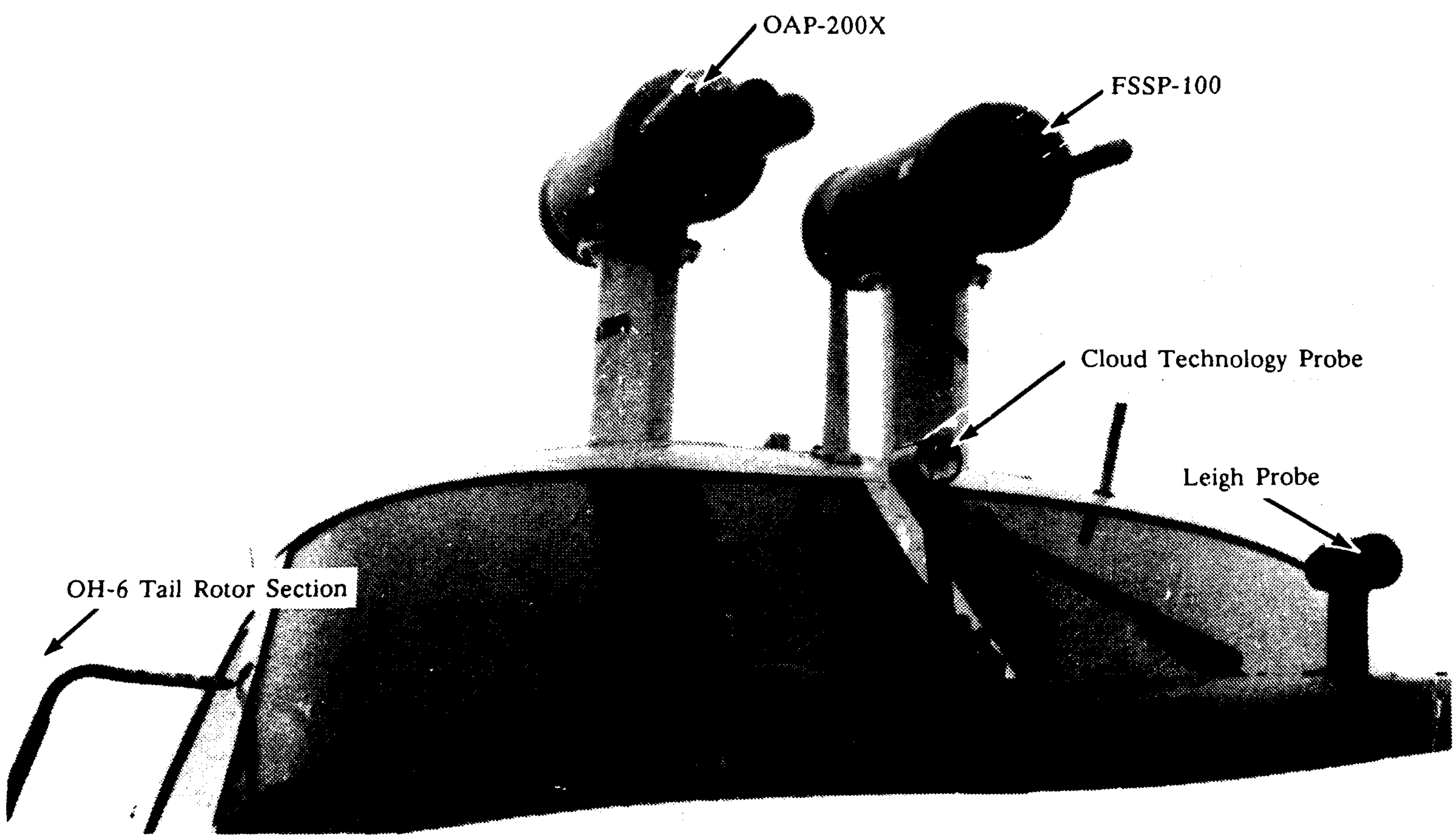

Figure C-1. JU-21A Aircraft – Nose and Cabin Top View

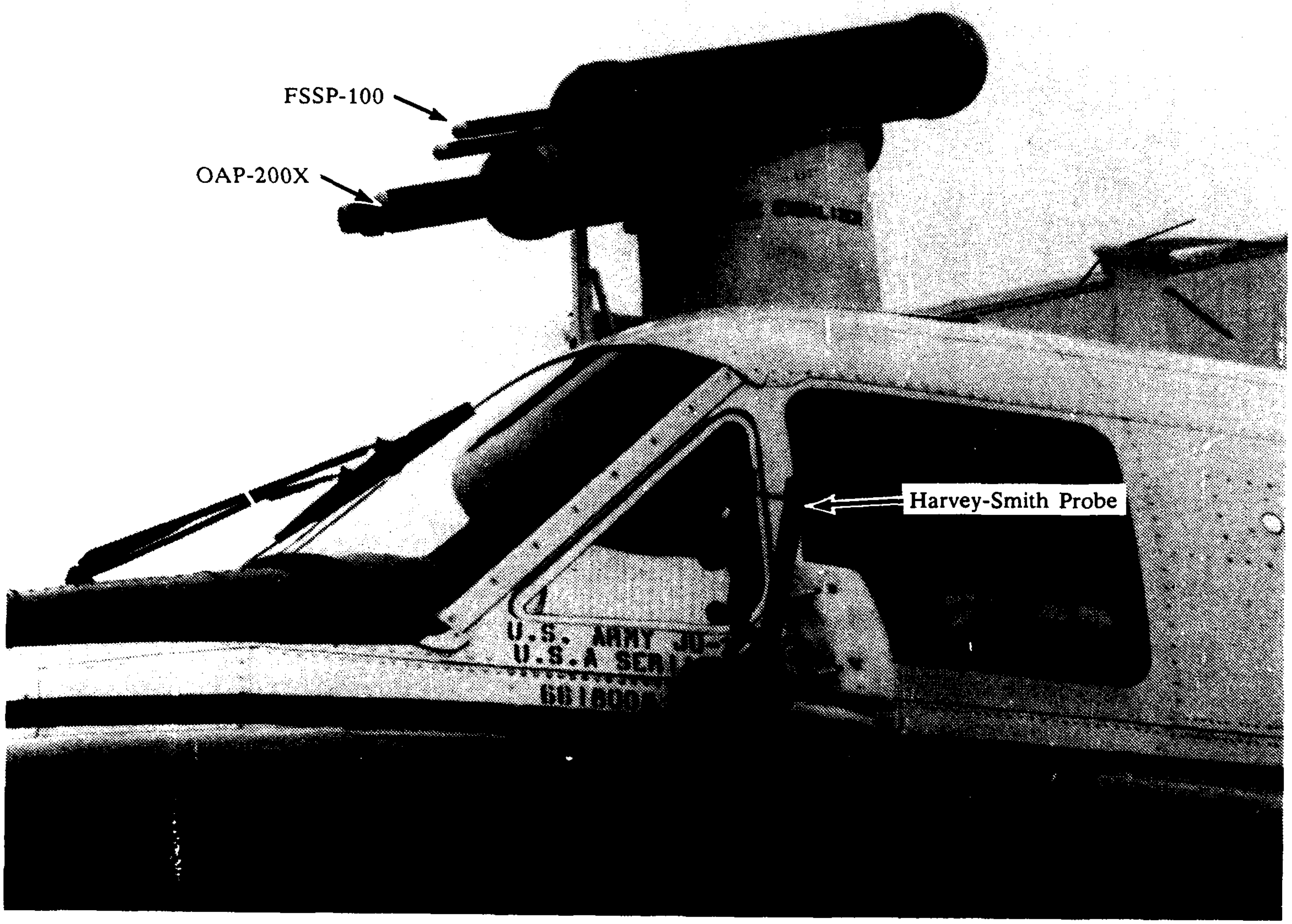

Figure C-2. JU-21 Aircraft – Left Side View

Figure C-3. JU-21A Aircraft - Interior Instrumentation Rack

5. The OAP-200X determines particle size using a linear array of photodiodes to sense the shadowing of array elements. Particles passing through the field of view illuminated by its laser are imaged as shadowgraphs on the array and a flip-flop memory element is set if the photodiode elements are darkened. Size is given by the number of elements set by a particle's passage, the size of each array element, and the optical magnification. Magnification is set for a size range of 20 to 300 µm and 24 active photodiode element's divide particles into 15 size channels, each 20 µm wide. It is capable of sizing particles with velocities of 5 to 100 meter/sec (10 to 194 knots). Depth of field, effective array width, and sample area vary with sensed particle size to a maximum of 61 mm, 0.44mm and 18.3 mm^2, respectively.

6. The SIIDS was designed by Meteorological Research Inc. and is a data acquisition system programmed specifically for icing studies. A more complete description appears in reference 9, appendix A. It consists of four main components: a microprocessor, Techtran data cassette recorder, Axiom printer, and an operator control panel. The SIIDS has three operational modes: (1) data acquisition, in which averaged raw data are recorded on cassette tape and engineering units are displayed on the printer, (2) a playback mode in which raw averaged data read from the cassette are converted to engineering units displayed on the printer, and (3) a monitor mode used to set the calendar clock and alter programmed constants. During data acquisitions, the operator may select an averaging period of 1/2, 1, 2 ,5, or 10 seconds. The following parameters are displayed on the SIIDS printer in engineering units.

a. calendar: year, month, day, hour, minute and second
b. pressure altitude (feet)
c. airspeed (knots)
d. outside air temperature (degree C)
e. dew point (degree C)
f. total LWC observed by the FSSP (gm/m^3)
g. total LWC observed by summing both FSSP AND OAP (gm/m^3)
h. median volumetric diameter (µm)
i. amount of LWC observed for each channel (TOTAL 30) of both probes (gm/m^3)

7. The Cloud Technology ice detector (model LWH-1) has a calibrated resistance wire which is mounted in the airstream and connected as one branch of a balanced bridge circuit. The wire is heated by an electric current. As the water droplets in the cloud strike the wire, they are heated by an electric current. As the water droplets in the cloud strike the wire, they are evaporated, cooling the wire and decreasing its resistance. The change in resistance causes the bridge to become unbalanced. The degree of unbalance is a function of the LWC of the cloud. A second resistance wire, mounted with its axis parallel to the airstream direction and hence not subject to water-drop impingement, is connected as an adjacent branch of the bridge. This wire serves to compensate for variations in airspeed, altitude, and air temperature so that the bridge becomes unbalanced only in the presence of water droplets. The output of the bridge is proportional to the rate of impingement of water on the sensing wire. This signal is converted to concentration of water per unit volume of air by means of an adjustment for true airspeed.

DEFINITIONS

8. The following definitions are used in this report:

Deficiency: A defect or malfunction discovered during the life cycle of an item of equipment that constitutes a safety hazard to personnel; will result in serious damage to the equipment if operation is continued or indicates improper design or other cause of an item or part, which seriously impairs the equipment's operational capability.

Shortcoming: An imperfection or malfunction occurring during the life cycle of equipment, which must be reported and which should be corrected to increase efficiency and to render the equipment completely serviceable. It will not cause an immediate breakdown, jeopardize safe operation or materially reduce the usability of the material or end product.

Anti-ice System: Ice is prevented from forming on the surfaces which are continually heated.

Deice System: Ice is permitted to form on a surface which is then removed by a cycled deice system.

Icing Conditions: When ambient temperatures are +4 degrees C or below and visible liquid moisture is present, icing severity is defined by the liquid water content (LWC) of the outside air and measured in grams per cubic meter (g/m3).

a. Trace: LWC 0 TO 0.15 g/m^3.
b. Light: LWC 0.15 to 0.5 g/m^3.
c. Moderate: LWC O.5 to 1.0 g/m^3.
d. Heavy: LWC greater than 1.0 g/m^3.

APPENDIX D. PHOTOGRAPHS

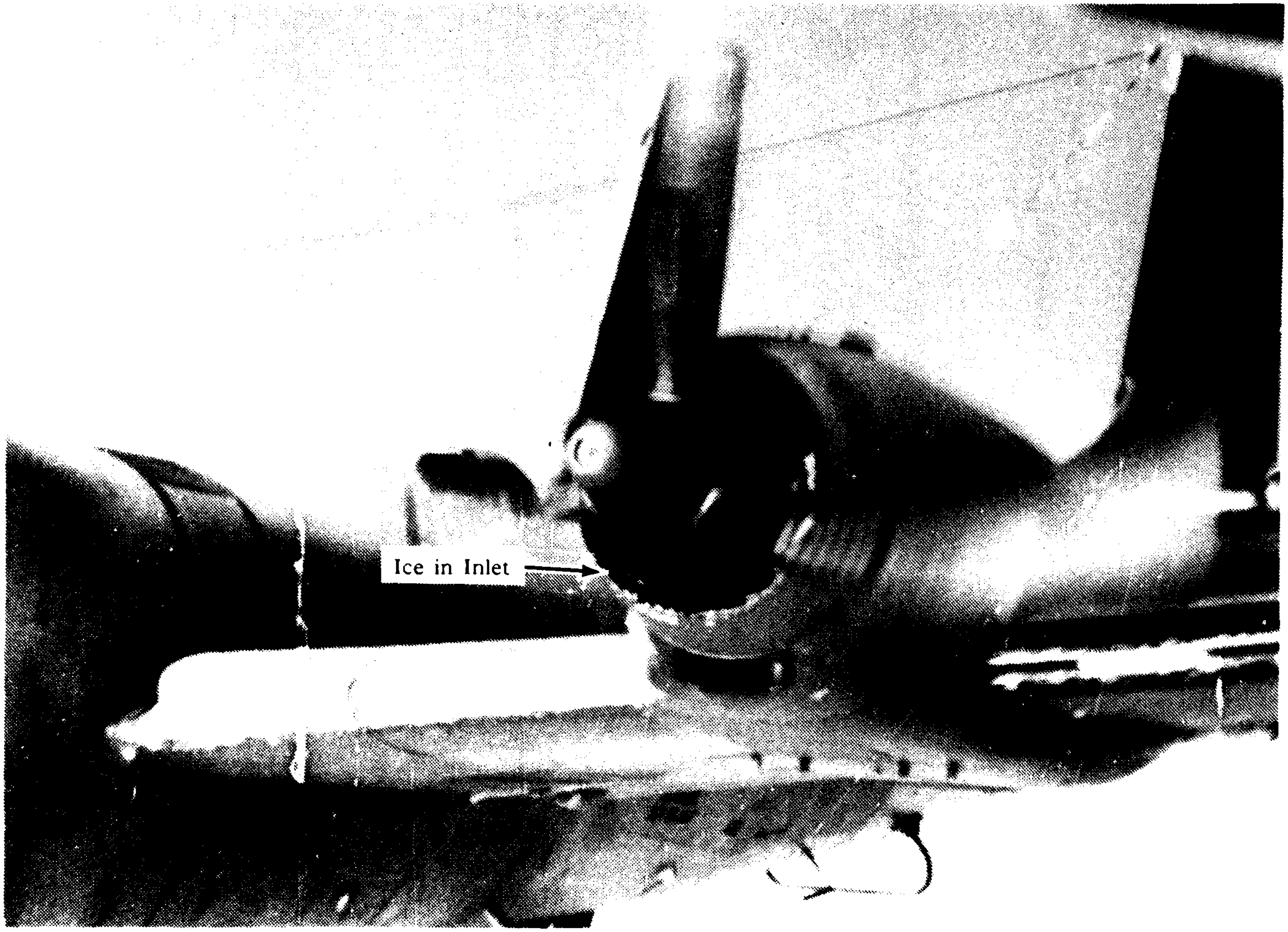

Figure D-1. Ice in Inlet: LWC = 0.6 gm/m^3, Time in Cloud = 0.5 hr, OAT = -11°C

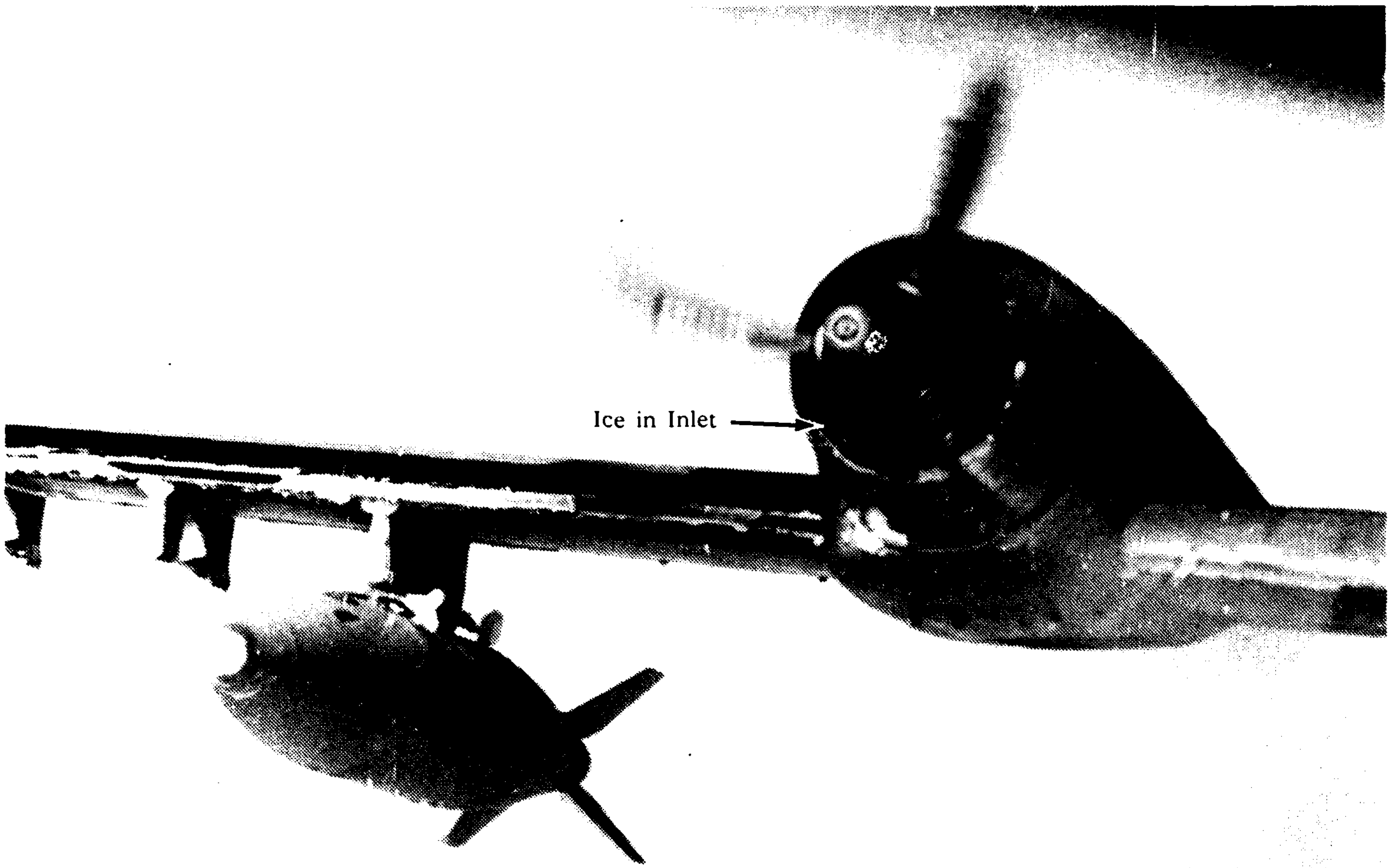

Figure D-2. Ice in Inlet: LWC = 0.6 gm/m³, Time in Cloud = 0.5 hr, OAT = -11°C

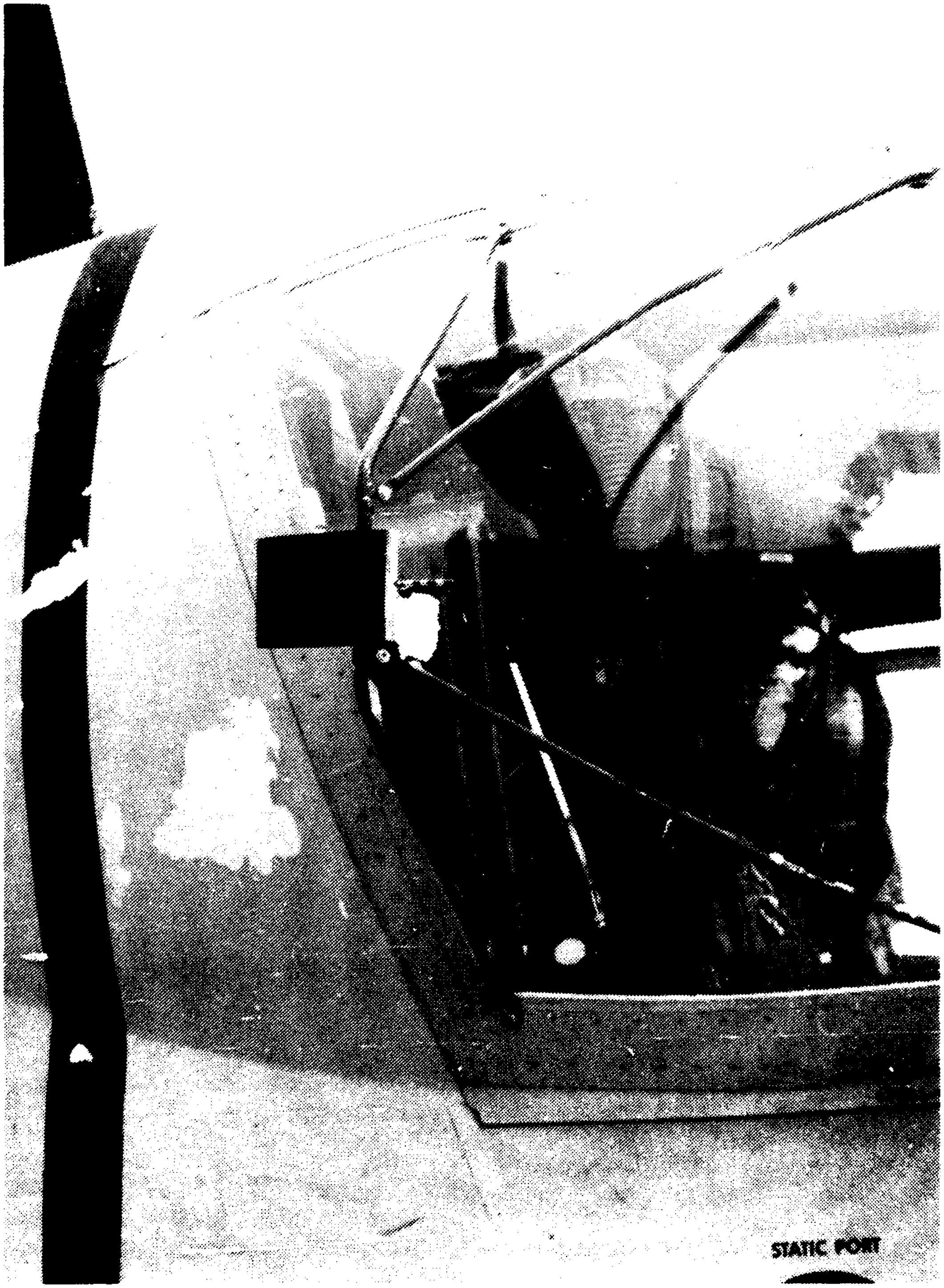

Figure D-3. Ice Damage from Propeller

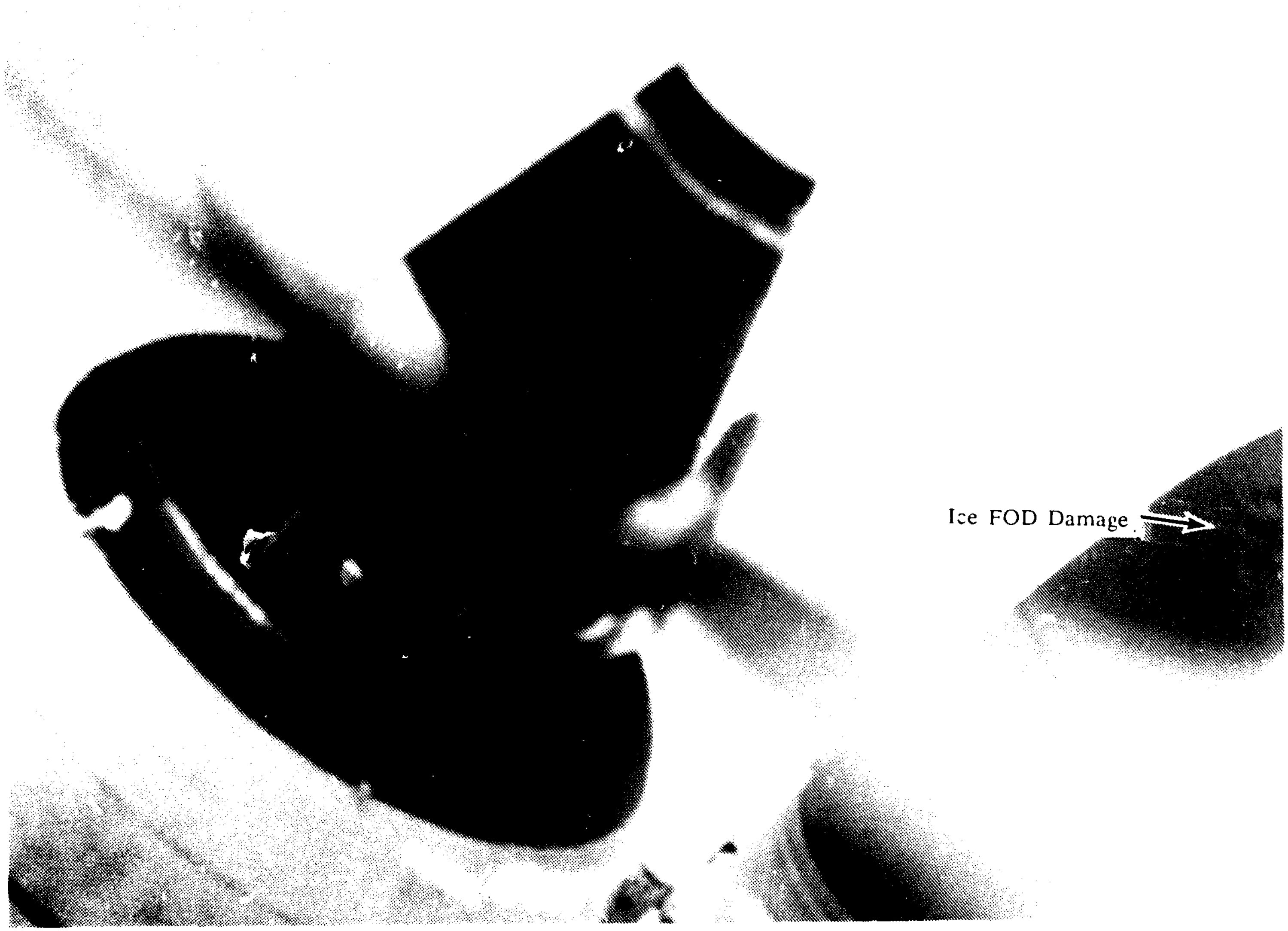

Figure D-4. Ice Damage to Left Drop Tank from Propeller

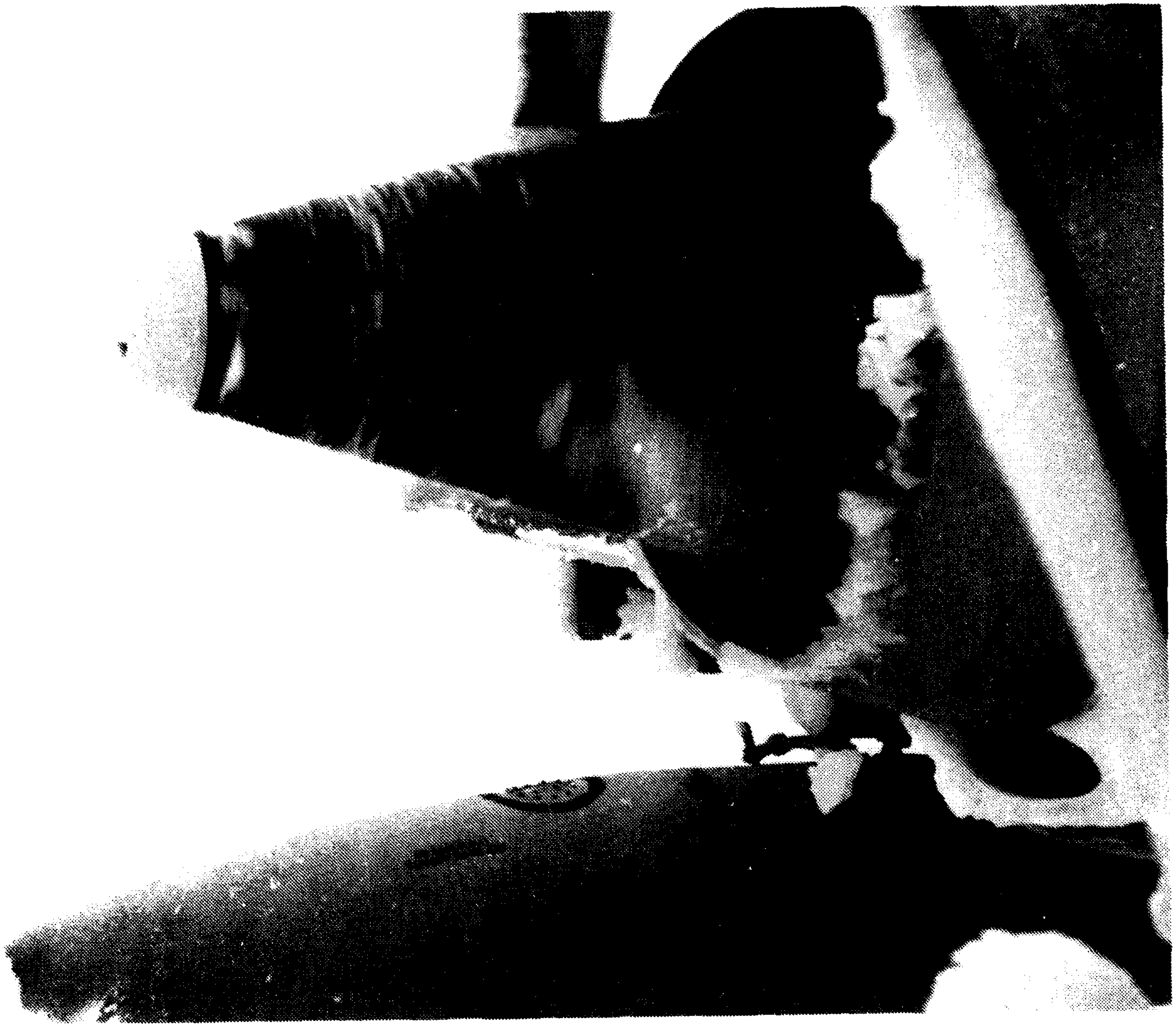

Figure D-5. Propeller, Nose, Spinner Ice Accretions: LWC = 0.6 gm/m^3, Time in Cloud = 0.5 hr, OAT = -11°C

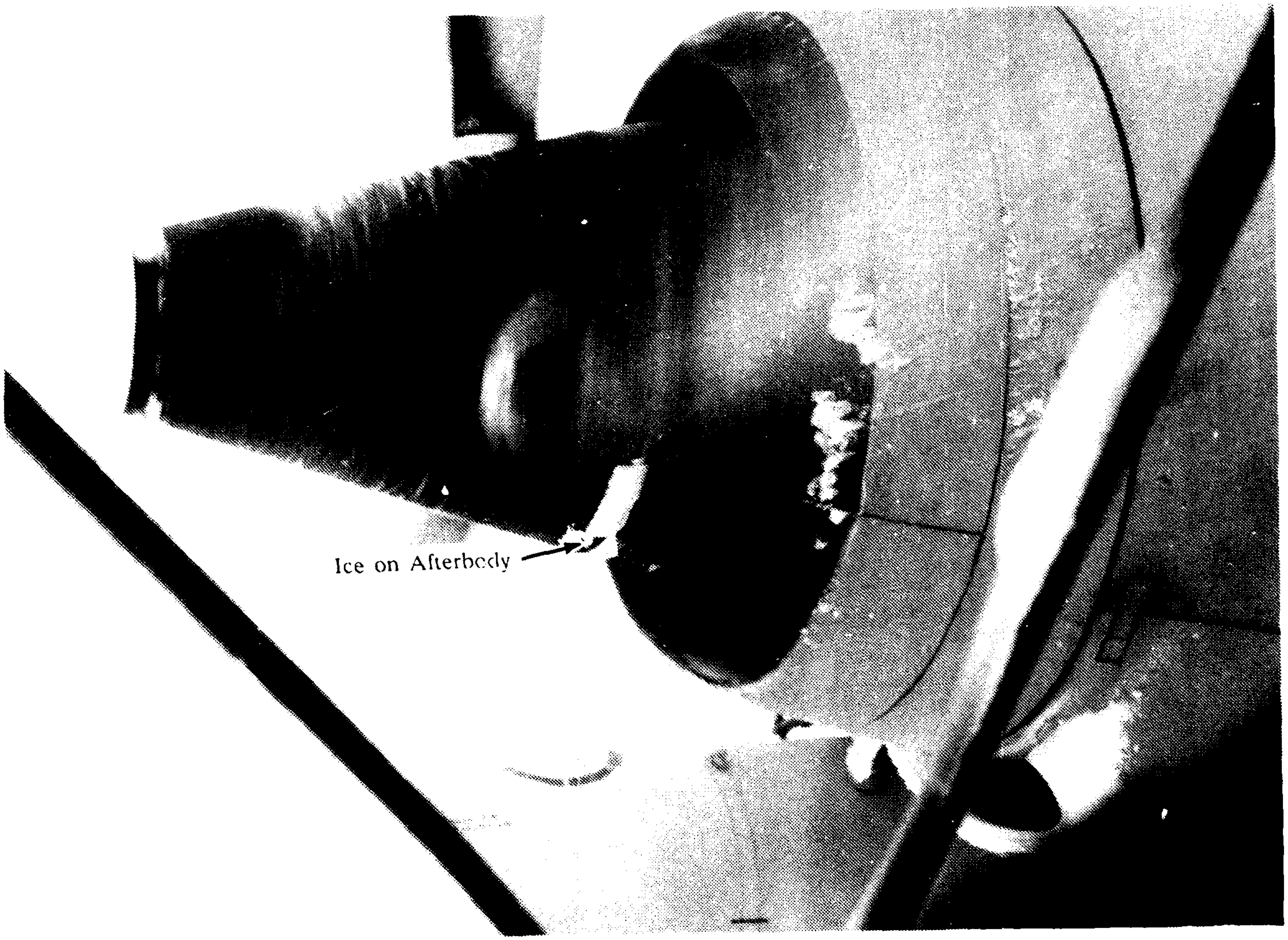

Figure D-6. Propeller Spinner Afterbody: LWC = 0.6 gm/m³, Time in Cloud = 0.8 hr, OAT = -8.0°C

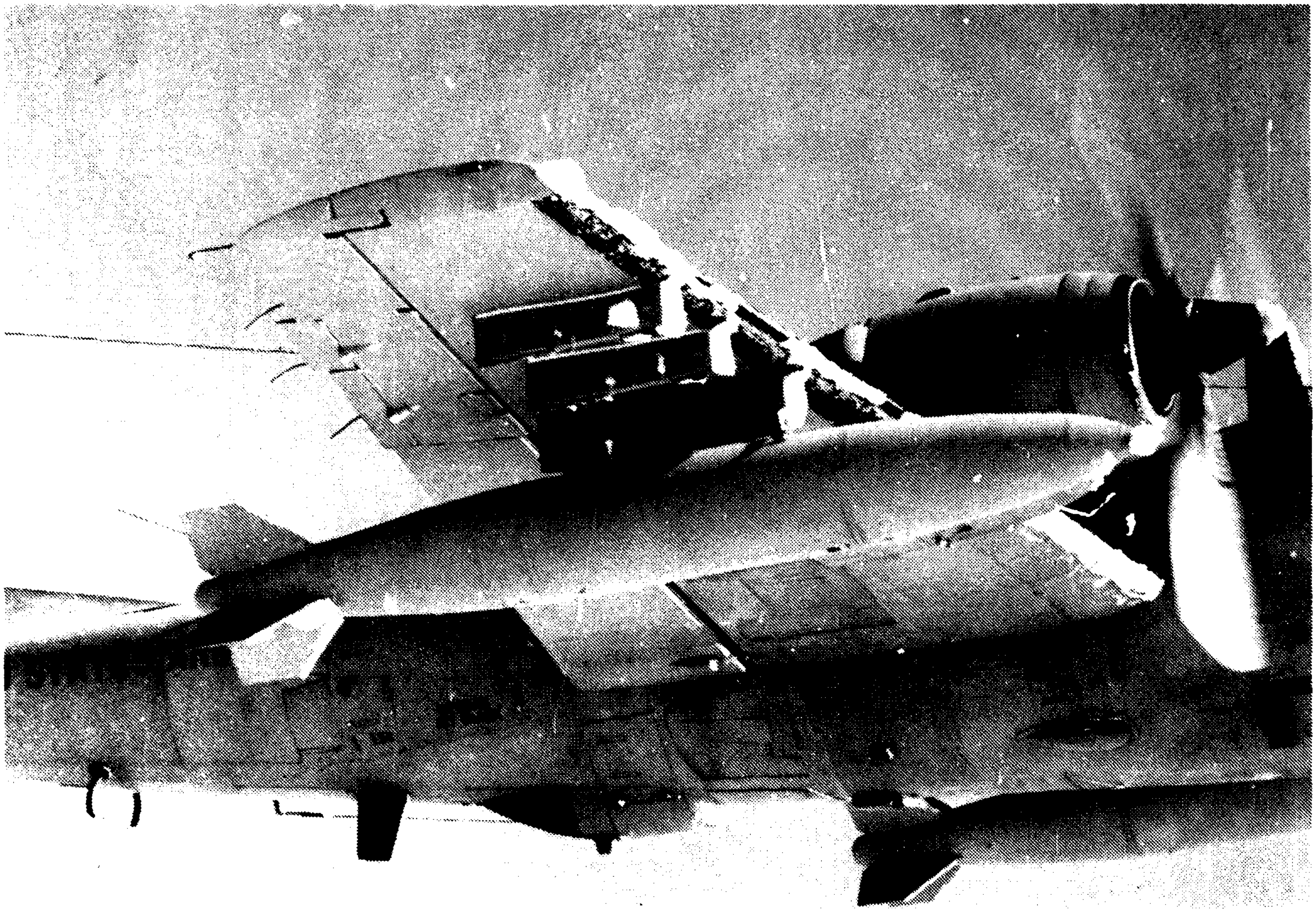

Figure D-7. Wing/Vertical/Horizontal Stabilizer Leading Edge Ice Accretions: LWC = 0.17 gm/m^3, Time in Cloud = 1.0 hr, OAT = -12°C

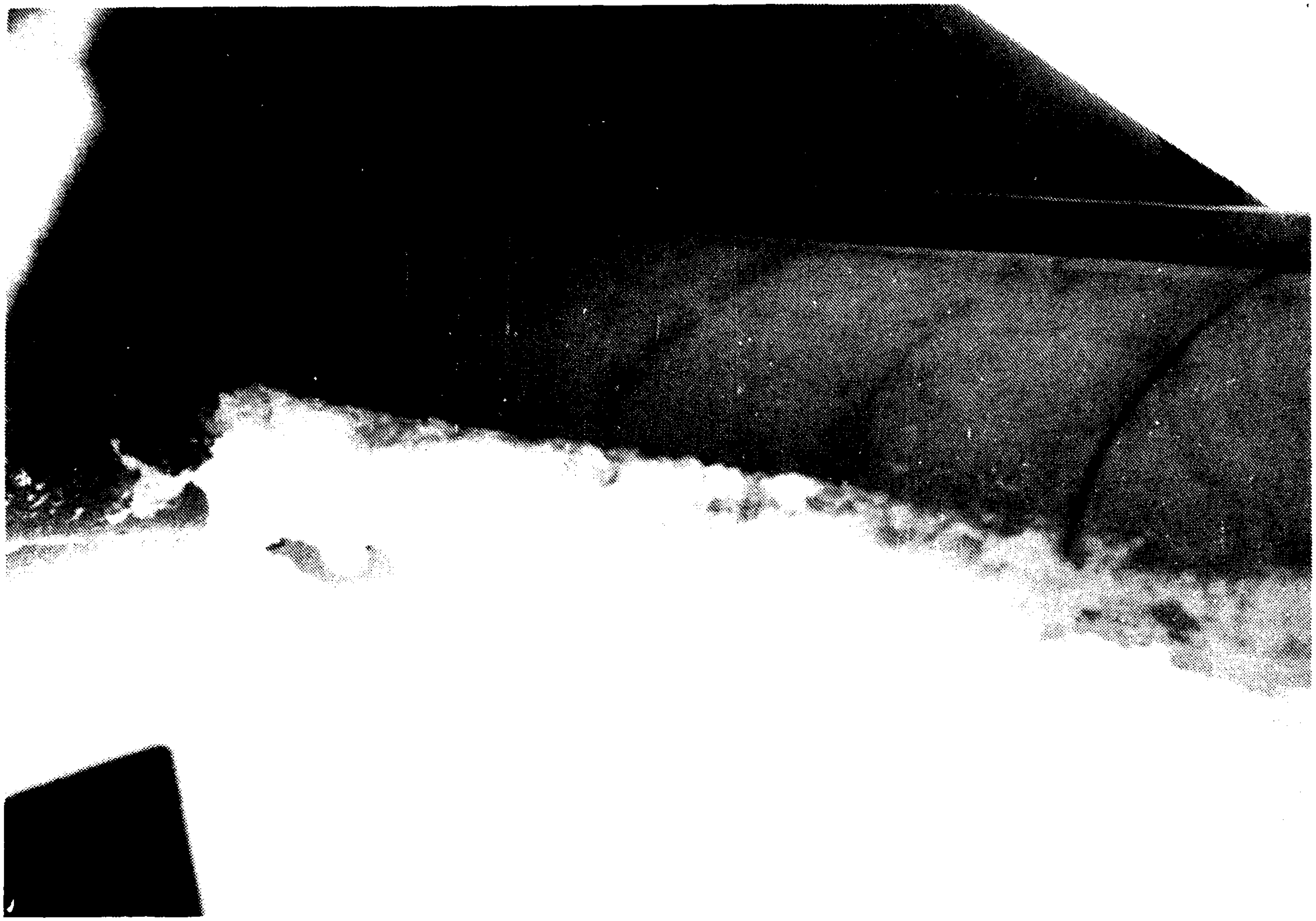

Figure D-8. Inboard Wing Section Ice: LWC = 0.3 gm/m³, Time in Cloud = 0.8 hr, OAT = 8.5°C

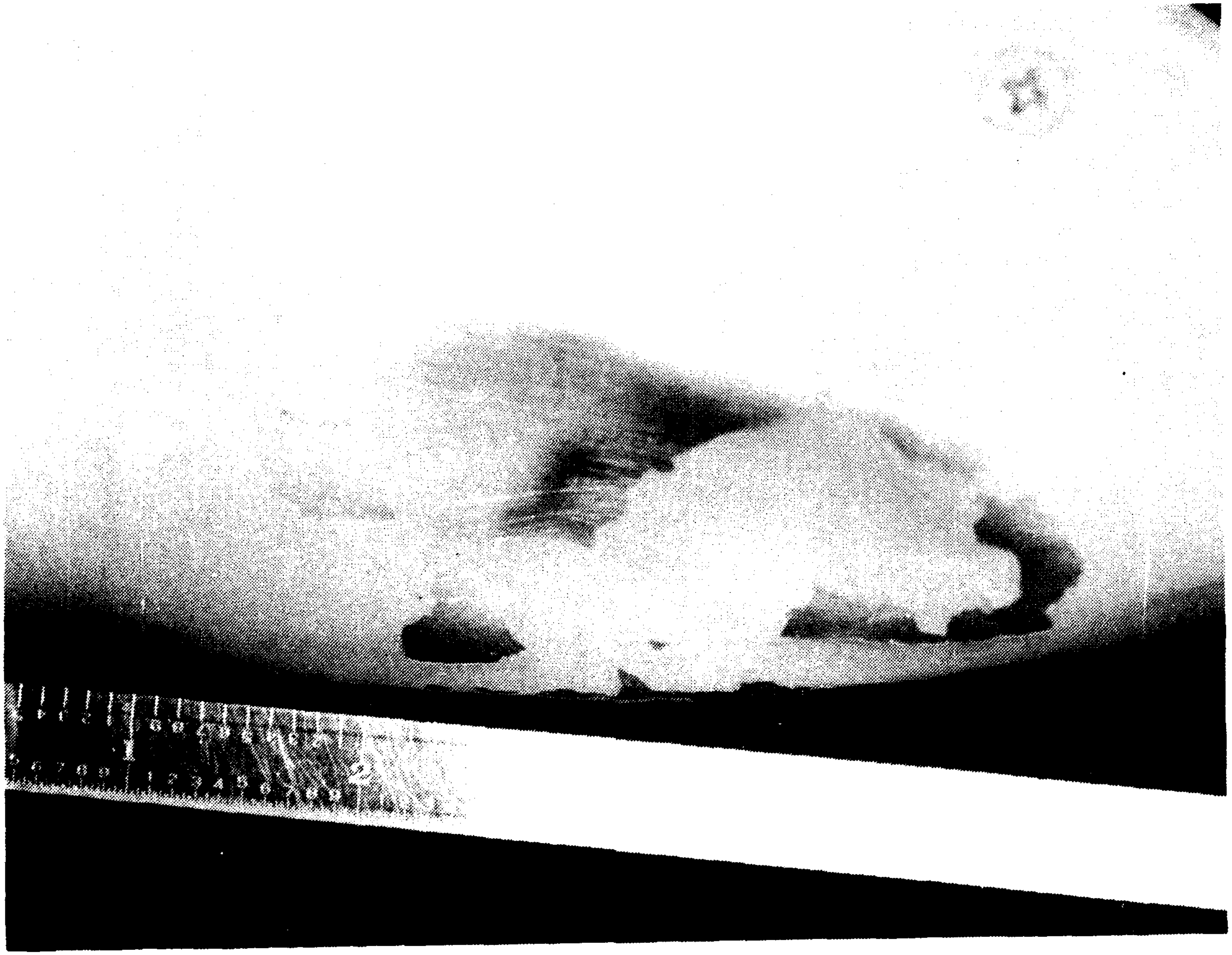

Figure D-9. Ice Damage to Vertical Stabilizer

Figure D-10. Windshield/Vertcal Stabilizer Ice: LWC = 0.6 gm/m^3, Time in Cloud = 0.9 hr, OAT = -8.0°C

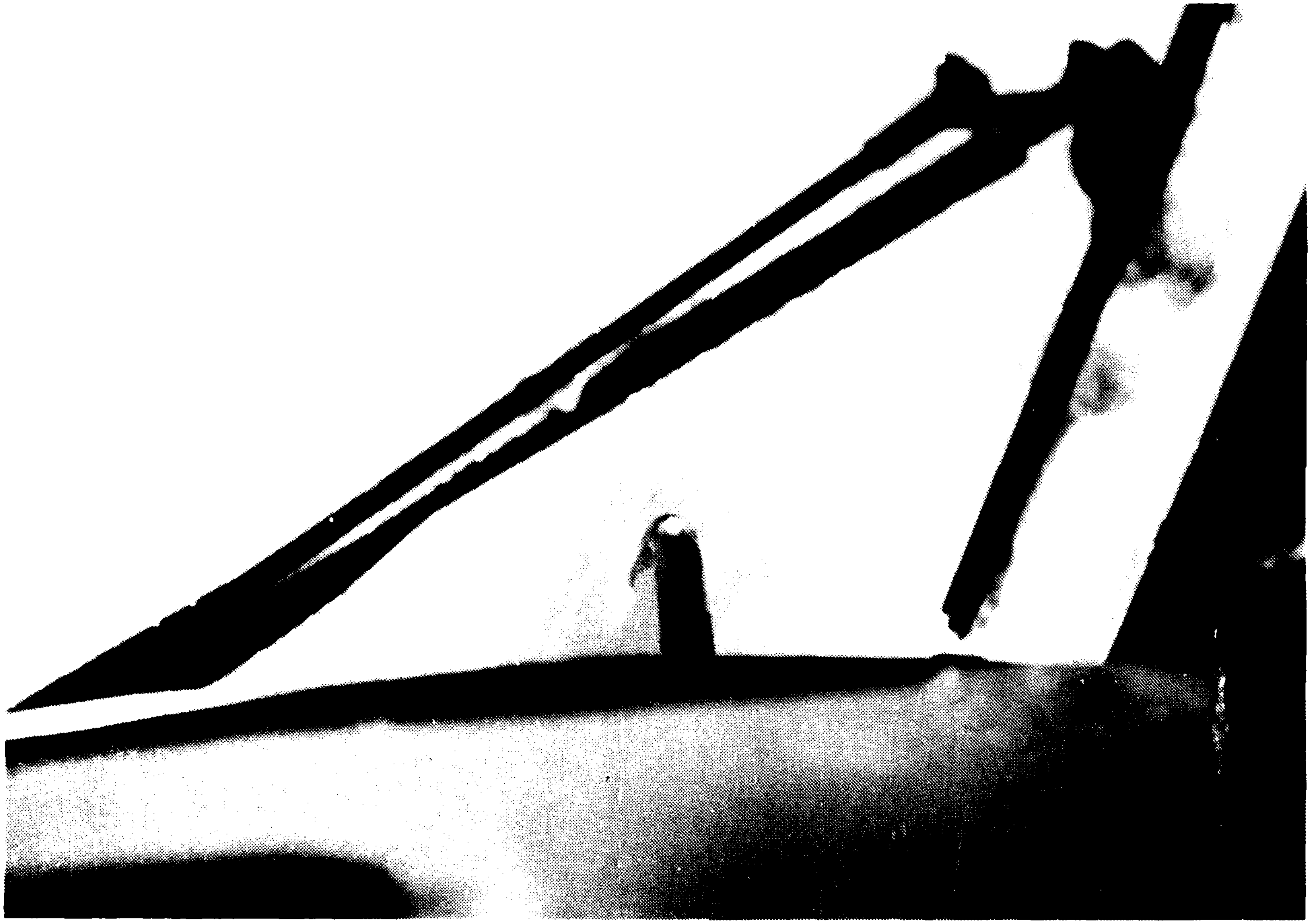

Figure D-11. Ice on Pitot Tube: LWC = 0.6 g/m^3, Time in Cloud = 0.9 hr, OAT = -8.0°C

Figure D-12. Ice on Pitot Tube: LWC = 0.6 g/m³, Time in Cloud = 0.9 hr, OAT = -8.0°C

APPENDIX E. TABLES

Table E-1. Engine Acceleration Tests 5000' Hp[1]

Test Engine	ECU Bleed Air	AC GEN Config (L/R)	Starting % N1	Max EGT (°C)	Time (Sec)
Left S/N LE-01414AX TSO 30 TSN 2655	Off	On/On	85	525	3
			80	540	3
			75	540	3
			70	543	4
		On/Backup	85	560	3
			80	558	3
			75	558	4
			70	562	5.5
	On	On/On	85	555	3
			80	550	4
			75	552	4
			70	560	5
		On/Backup	85	563	4
			80	561	4
			75	575	6
			70	585	6
Right S/N LE-30015A TSO 1590 TSN 3070	Off	On/On	85	530	4
			80	535	5
			75	537	5
			70	565	5
		Backup/On	85	512	3.5
			80	540	3.5
			75	561	4
			70	589	5
	On	On/On	85	520	4
			80	535	4.5
			75	561	4.5
			70	571	5
		Backup/On	85	532	4
			80	552	4
			75	550	4
			70	583	5

Remarks: OAT (°C) -9 Airspeed 135 KIAS

NOTE:

[1]No mission equipment other than the aircraft anti-ice/deice equipment was being powered. MAX EGT and TIME repesents values observed from starting N1 speed to maximum N1 speed obtainable at full throttle.

Table E-2. Engine Acceleration Tests 10,000' Hp[1]

Test Engine	ECU Bleed Air	AC GEN Config (L/R)	Starting % N1	Max EGT (°C)	Time (Sec)
Left S/N LE-01414AX TSO 30 TSN 2655	Off	On/On	85	533	4
			80	526	5
			75	529	6
			70	539	7
		On/Backup	85	512	3.5
			80	546	6
			75	561	6
			70	568	7
	On	On/On	85	534	5
			80	535	6
			75	547	6
			70	550	7
		On/Backup	85	562	5
			80	556	6
			75	570	7
			70	580	8
Right S/N LE-30015A TSO 1590 TSN 3070	Off	On/On	85	492	4.5
			80	523	5
			75	530	5
			70	553	5
		Backup/On	85	491	4
			80	533	4
			75	576	5
			70	586	5.5
	On	On/On	85	498	4.5
			80	535	4.5
			75	543	5
			70	575	5.5
		Backup/On	85	524	4
			80	548	4.5
			75	580	4.5
			70	599	6

Remarks: OAT (°C) -20 Airspeed 135 KIAS

NOTE:

[1]No mission equipment other than the aircraft anti-ice/deice equipment was being powered. MAX EGT and TIME repesents values observed from starting N1 speed to maximum N1 speed obtainable at full throttle.

Table E-3. Engine Acceleration Tests 15,000' Hp[1]

Test Engine	ECU Bleed Air	AC GEN Config (L/R)	Starting % N1	Max EGT (°C)	Time (Sec)
Left S/N LE-01414AX TSO 30 TSN 2655	Off	On/On	85	522	5
			80	522	5
			75	526	6
			70	526	7.5
		On/Backup	85	534	5
			80	558	5.5
			75	535	6
			70	560	7.5
	On	On/On	85	530	4
			80	538	5
			75	543	6
			70	557	7.5
		On/Backup	85	558	5
			80	564	6.5
			75	565	7.5
			70	575	9
Right S/N LE-30015A TSO 1590 TSN 3070	Off	On/On	85	500	4
			80	525	4.5
			75	543	4.5
			70	561	6
		Backup/On	85	520	4.5
			80	538	4.5
			75	574	5
			70	581	6
	On	On/On	85	516	5
			80	534	5
			75	538	5
			70	575	6
		Backup/On	85	521	3.5
			80	542	4.5
			75	589	5
			70	594	6

Remarks: OAT (°C) -26 Airspeed 135 KIAS

NOTE:

[1]No mission equipment other than the aircraft anti-ice/deice equipment was being powered. MAX EGT and TIME repesents values observed from starting N1 speed to maximum N1 speed obtainable at full throttle.

Table E-4. Engine Acceleration Tests 20,000' Hp[1]

Test Engine	ECU Bleed Air	AC GEN Config (L/R)	Starting % N1	Max EGT (°C)	Time (Sec)
Left S/N LE-01414AX TSO 30 TSN 2655	Off	On/On	85	543	5
			80	549	6
			75	524	7.5
			70	532	9
		On/Backup	85	550	6
			80	558	7
			75	551	9
			70	565	12.0
	On	On/On	85	545	5.5
			80	537	6
			75	563	7.5
			70	553	8.5
		On/Backup	85	570	6
			80	566	8
			75	590	10
			70	603	12.5
Right S/N LE-30015A TSO 1590 TSN 3070	Off	On/On	85	510	4
			80	513	5
			75	541	5.5
			70	590	6
		Backup/On	85	552	4
			80	573	5
			75	589	6
			70	601	7
	On	On/On	85	523	4
			80	532	5.5
			75	580	5.5
			70	584	7
		Backup/On	85	547	4.5
			80	555	5
			75	595	6.5
			70	624	7

Remarks: OAT (°C) -34 Airspeed 130 KIAS

NOTE:

[1]No mission equipment other than the aircraft anti-ice/deice equipment was being powered. MAX EGT and TIME repesents values observed from starting N1 speed to maximum N1 speed obtainable at full throttle.

Table E-5. Engine Acceleration Tests 25,000' Hp[1]

Test Engine	ECU Bleed Air	AC GEN Config (L/R)	Starting % N1	Max EGT (°C)	Time (Sec)
Left S/N LE-01414AX TSO 30 TSN 2655	Off	On/On	85	572	6
			80	580	7
			75	554	8
			70	570	11
		On/Backup	85	570	8
			80	567	9
			75	594	13
			70	625	16
	On	On/On	85	558	6
			80	575	7
			75	583	9
			70	550	11
		On/Backup	85	593	6
			80	578	9
			75	610	11
			70	625	17
Right S/N LE-30015A TSO 1590 TSN 3070	Off	On/On	85	537	4
			80	534	5
			75	570	6
			70	594	8
		Backup/On	85	544	5
			80	545	6
			75	592	7.5
			70	635	9
	On	On/On	85	526	4
			80	540	5
			75	595	6
			70	595	8.5
		Backup/On	85	551	5
			80	558	6
			75	618	7
			70	646	10

Remarks: OAT (°C) -38 Airspeed 130 KIAS

NOTE:

[1]No mission equipment other than the aircraft anti-ice/deice equipment was being powered. MAX EGT and TIME repesents values observed from starting N1 speed to maximum N1 speed obtainable at full throttle.

APPENDIX F. TEST INCIDENT REPORTS

TEST INCIDENT REPORT (AMCR 70-13)

1. Release Date: 21 March 1989

2. Test Title: Evaluation of the Improved OV-1D Anti-ice System

3. Test Project #: 87-25-1

4. TIR #: EC-B87251002

5. Test Agency: AEFA

6. Test Sponsor AVSCOM

I MAJOR ITEM DATA

10. Model: ASH 811-1 10KVA Converter
11. Serial#: +D504
12. USA#: NA
13. Mfr: Leland
14. Contract#: DAAK50-83-G-0004-001

Test Life: Units:

21.
22. UNKNOWN
23.
24. (Not Used)

II INCIDENT DATA

30. Title Converter Failure/Dropout
31. Subsystem: OV-1D
32. Incident Class: Minor
33. Category: RAM
34. Observed During: Operation
35. Action Taken: None

40. Data & Time: 6 March 1989, 1130 CST
41. FD/SC Step#: NA
42. FD/SC Class: NA
43. Chargeability: Hardware
44. Preliminary CA Status: Open
45. Asgd Resp: Materiel Developer

48. Test Environment: Stall Surge Test at 5000 feet pressure altitude
49. Defective Materiel: Returned to Grumman/Leland

III INCIDENT SUBJECT DATA

50. Name AC Converter
51. Serial#: UNK
52. FSN/NSN: NA
53. Mfr: Leland
54. Mfr Part#: ASH 811-1
55. Drawing#: NA
56. Quantity: One
57. Action: Replaced

60. FGC: OV-1D 68-15934
61. LSA#: NA

Part Life: Units:

63. NA
64. NA
65. NA
66. Next Assy: NA
67. Serial#: NA

IV MAINTENANCE DATA

70. Diagnostic Clockhours: 0.5
71. Diagnostic Manhours: 1.0
72. Active Maint Clockhours: 0.5
73. Active Maint Manhours: 1.0

80. Type: Unscheduled
81. Level Use: AVUM
82. Level Prsc: None
83. Level Recm: AVUM

V INCIDENT DESCRIPTION

Full Description of Incident:

90. Upon completion of stall surge tests at 5000 feet, the anti-ice systems were turned off, the #1 AC generator turned on and then the #1 converter was switched on. The #1 converter would not reset after several attempts and a slight electrical insulation fumes were noticed. Troubleshooting indicates the #1 converter has only 1 phase operational.

Name, Title & Phone of Preparer:

98. JOSEPH C. MIESS, CW4
Project Officer, AUTOVON 527-4986

FOR THE COMMANDER:

99. PAUL W. LOSIER, MAJ, AV
Chief, Plans & Programs

TEST INCIDENT REPORT (AMCR 70-13)	1. Release Date: 10 March 1989	
2. Test Title: Evaluation of the Improved OV-1D Anti-ice System	3. Test Project # 87-25-1	4. TIR # EC-B87-25-1001
5. Test Agency: AEFA	6. Test Sponsor AVSCOM	

I MAJOR ITEM DATA

10. Model: DT-5041A Ice Detector	Test Life: Units:
11. Serial#: DAA 259	21.
12. USA#: NA	22. UNKNOWN
13. Mfr: Rosemount	23.
14. Contract#: DAAJ09-85-G-A030	24. (Not Used)

II INCIDENT DATA

30. Title Ice Detector Failure	40. Data & Time: 7 March 1989, 1000 CST
31. Subsystem: OV-1D	41. FD/SC Step#: NA
32. Incident Class: MINOR	42. FD/SC Class: NA
33. Category: RAM	43. Chargeability: Hardware
34. Observed During: Operation	44. Preliminary CA Status: Open
35. Action Taken: Changed Ice Detector	45. Asgd Resp: Materiel Developer

48. Test Environment: Natural Icing Test

49. Defective Materiel: Returned to Rosemount

III INCIDENT SUBJECT DATA

50. Name Ice Detector	60. FGC: OV-1D 68-15934
51. Serial#: DAA 259	61. LSA#: NA
52. FSN/NSN: NA	Part Life: Units:
53. Mfr: Rosemount	63. NA
54. Mfr Part#: A51A9030-3	64. NA
55. Drawing#:	65. NA
56. Quantity: One	66. Next Assy: Anti-ice/De-ice Controllers
57. Action: Replaced	67. Serial#: NA

IV MAINTENANCE DATA

70. Diagnostic Clockhours: 0.5	80. Type: Unscheduled
71. Diagnostic Manhours: 0.5	81. Level Use: AVUM
72. Active Maint Clockhours: 1.0	82. Level Prsc: None
73. Active Maint Manhours: 1.0	83. Level Recm: AVUM

V INCIDENT DESCRIPTION

Full Description of Incident:

90. The engine anti-ice/de-ice system automatic turn-on stopped after 1.5 hours flight in very light icing. Ice accreted to approximately 3/8" on engine inlet cowl ring. The engine anti-ice switch was placed to emergency and all surfaces were free of ice in approximately 5 minutes.

Name, Title & Phone of Preparer:	FOR THE COMMANDER:
98. JOSEPH C. MIESS, CW4 Project Officer, AV 527-4786	99. PAUL W. LOSIER, ~~MAJ~~ AV Chief, Plans & Programs

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